JOURNAL OF THE AERONAUTICAL SCIENCES

VOLUME 13 FEBRUARY, 1946 NUMBER 2

British Aircraft Gas Turbines

The Ninth Wright Brothers Lecture*

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SUMMARY

This paper is an introduction to the work done in Great Britain on aircraft gas turbines. It is intended to give a general impression, leaving the wealth of detail to be supplied by others. It deals with the early work of Whittle, Griffith, and Constant, done before the war in circumstances of considerable difficulty. The two main lines of development associated with axial and centrifugal compressors are traced, and the extension of this work into practically the whole of the British aircraft engine industry is described. The contributions of the various British firms are noticed, and the collaboration which was developed between them is the subject of special comment. There follow notes on the parallel collaboration which was developed between Great Britain and the United States and between Great Britain and the British Dominions.

Some of the problems encountered in development and in the research which was always behind the development are dealt with, and the facilities which are at the service of the British gas turbine engineers are described. The paper contains some discussion of the variety of problems which the advent of the aircraft gas turbine has initiated; for example, the characteristics of single- and double-sided centrifugal compressors are compared; the merits of axial and centrifugal compressors, of jet reaction, ducted fan, and propeller propulsive systems are considered; and some attempt is made to deal with the influences of the gas turbine on air-frame design.

1. INTRODUCTION

FIRST, I WISH TO EXPRESS to the Institute of the Aeronautical Sciences my sincere appreciation of their invitation to give this lecture. To be given the opportunity of paying tribute in this way to the Wright Brothers in their native land is an honor of which I am deeply sensible, and it is in a spirit of homage to two great Americans that I offer this account of part of what I believe to be the greatest aeronautical revolution since the Wright Brothers made history 42 years ago.

In speaking of a revolution I am referring to the major change which the gas turbine is making in the design and performance of the airplane. Like most great technical advances, it is due to no one country and no one man. Restricting consideration to the aircraft gas turbine, the major parts have been played by Great Britain, America, and Germany, and one day a history in which their pioneering contributions will be properly described and assessed will, I hope, be written. We are, though, still too near to the technical works which such a history will review for it to be written now in correct perspective. But there is no doubt that the time has come for those who have been concerned in aircraft gas turbine work to give accounts of the parts of the work with which they are most familiar. The restrictions which, for reasons of military security, prevented the publication of such accounts a few months ago are now sufficiently relaxed for useful beginnings in the dissemination of the tremendous volume of material which has accumulated in the past few years to be made.

* Presented before the Institute of the Aeronautical Sciences in the U.S. Chamber of Commerce Auditorium, Washington, D.C., December 17, 1945.

† Chairman and Managing Director.

I therefore welcomed the suggestion of your Institution that my discourse should be in connection with aircraft gas turbines. I could, at least, for reasons I will note later, give a general picture of the work that has been done in Britain. That is what I am attempting in this paper. I hope that an American who has been in a like position to see his country's gas turbine work will paint a similar general picture of American work. Possibly there will be found a German to perform a corresponding service for the German work.

Such pictures should provide a background against which the more detailed studies that are beginning to appear may be better appreciated and, in some measure, related. The problem in painting them is to achieve breadth without excessive length. If my pursuit of that policy results in some blank patches, and in parts being only lightly touched in, I trust I may be forgiven.

I hope you will not consider a short personal note out of place. My own part in the gas turbine story has been that of the administrator and coordinator rather than that of the engineer. From June, 1940, to the end of April, 1944, it was my good fortune to be the officer of the British Ministry of Aircraft Production primarily concerned with aircraft gas turbine research and development, beginning as Deputy Director of Scientific Research when Dr. Pye, well known to many of you and an Honorary Fellow of this Institution, was Director, and ending with the somewhat obscure title of Director of Special Projects. There is no difficulty now in guessing what the special projects were. At the beginning of May, 1944, in order to create a national gas turbine organization, the British Government bought the business of the pioneer firm, Power Jets Limited, joined to it the gas turbine laboratories which were at the time part of the Royal Aircraft Establishment, and placed the whole under the direction of a new Company, called Power Jets (Research and Development) Limited, in which the Government held all the shares. I became Chairman and Managing Director of this company, and in those capacities I have been privileged to continue my association with the aircraft gas turbine.

2. The Early Days

2.1 Origins

The first gas turbine patent was granted in England to John Barber in 1791, and since then there have been numerous gas turbine inventions. These have been adequately described elsewhere,[1, 2] and I shall concern myself only with the developments which have led directly to recent British achievements.

The present flow of British aircraft gas turbine work springs from two distinct streams, both with their beginnings in the twenties. One stream, owing its origin to Dr. A. A. Griffith, is associated with research into axial compressors. The other, which Air Commodore Frank Whittle has made his own, is associated with the centrifugal compressor. I propose briefly to trace these two streams from their sources to the present time.

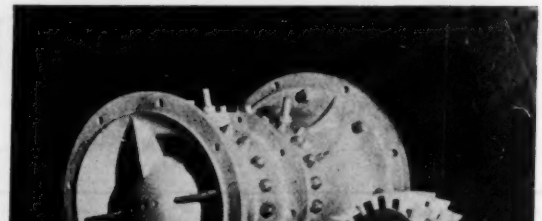

Fig. 1. Model turbocompressor tested at the Royal Aircraft Establishment in 1929.

2.2 Axial Compressors

It was in 1926 that A. A. Griffith produced, at the Royal Aircraft Establishment, an aerodynamic theory of turbine design based upon flow past airfoils as distinct from flows through passages. In October of that year he put his proposals before a meeting of representatives of the Air Ministry and the Aeronautical Research Committee. This meeting expressed itself in favor of preliminary investigations being done and, consequently, in 1927, two sets of experiments were put in hand.

One of these investigations produced the first Royal Aircraft Establishment tests, reported in 1928, on cascades of airfoils representing compressor and turbine blading.[3] The method of testing groups, or cascades of stationary blades in small wind tunnels designed for the purpose has since been developed and is widely used by gas turbine designers. Because of the absence of blade motion and, in many cascade tunnels, of the low air speed, the method is of value for qualitative and comparative purposes rather than for the determination of absolute blade criteria. Despite this obvious limitation, experience in the use of the cascade tunnel and correlation of its results with the results of turbine and compressor tests make it a useful weapon in the designer's armory.

The second set of experiments was done with a special test rig, designed during 1928, which consisted essentially of a single-stage turbine and a single-stage axial compressor mounted on the same shaft (Fig. 1). This rotor was driven by taking air through the casing by means of an exhauster pump, the air traveling past turbine stators, turbine rotating, compressor rotating, and compressor stator blades in that order. The blad-

ing had been designed on the basis of Griffith's theory. Testing began in January, 1929, and the results from this tiny rig—the internal casing diameter was 4 in. and the blade height approximately 0.5 in.—were extremely encouraging as an overall efficiency of nearly 91 per cent was obtained. There was no means of separating this into compressor and turbine efficiencies.

In November, 1929, Dr. Griffith discussed in an official paper the prospects of the internal combustion turbine driving an airscrew as an aircraft power plant. He concluded that it would be lighter, less bulky, and more efficient than piston engines projected at the time. The specific scheme he suggested was based on the contrarotating contraflow principle, and he proposed the construction of a rig to test it. A special panel of the Aeronautical Research Committee under the chairmanship of Sir Henry Tizard considered this paper and the other work which had been done on the gas turbine. While the panel did not advocate the construction of a turbine engine, they recommended in April, 1930, that the Griffith turbocompressor test rig be built for analyses of flow, pressure, and temperature distribution. They considered also that experiments on combustion of fuel at constant pressure should, if possible, be done as well.

If this work had been done as recommended, the axial compressor gas turbine in Britain might have been advanced some years. For reasons which are at this distance of time difficult to assess with certainty however, the proposed program was not put in hand. Further, interest in the axial compressor lapsed for 6 years. During this interregnum, the only work in the gas turbine field done at the Royal Aircraft Establishment was the continuation of experiments done in the late twenties on exhaust turbosuperchargers for piston engines. This work, though not directed to that end, provided knowledge of use in gas turbine development.

The second period of interest in axial compressors began in 1936. In July of that year the Royal Aircraft Establishment obtained authority to build an axial flow compressor. This unit had eight stages of free vortex blading and was christened "Anne" (Figs. 2 and 3). The diameter over the blading was 6 in. It was originally built with adjustable blowoff at each stage and on the delivery diffuser. The stage blowoff was to avoid blade stalling during starting, and the diffuser bleed was for boundary layer control. Work on this unit proceeded slowly and culminated in tragedy. For the unit was not completed until early in 1938 and stripped its blading 30 sec. after starting because of a faulty oil seal which caused rapid overheating.

The compressor was reconstructed without the blowoff arrangements and with larger clearances between the blade rows. These changes were made on the strength of Professor Ackeret's work in Switzerland (cf. § 5.5).

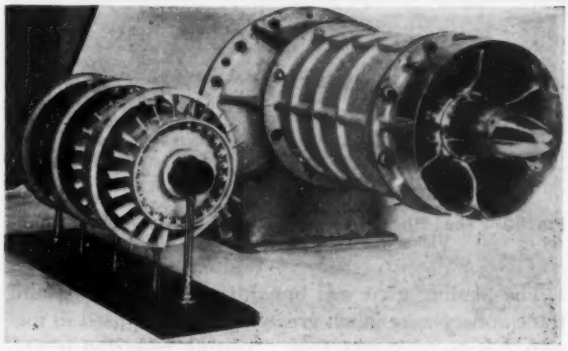

FIG. 2. The Royal Aircraft Establishment Axial Flow Compressor "Anne" (1938).

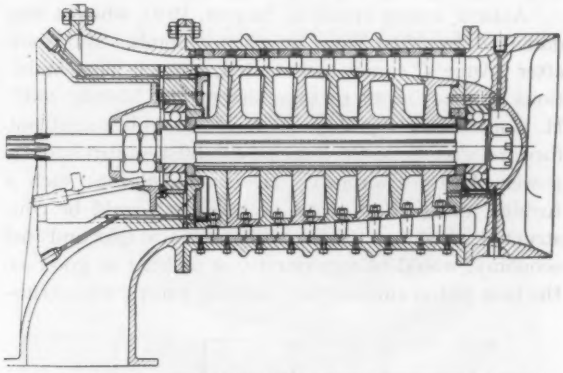

FIG. 3. Arrangement of the Axial Flow Compressor "Anne."

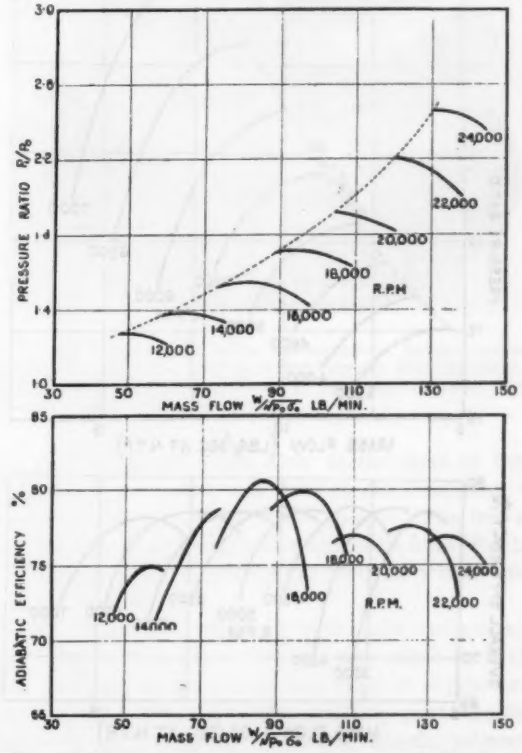

FIG. 4. Performance characteristics of the Axial Compressor "Anne."

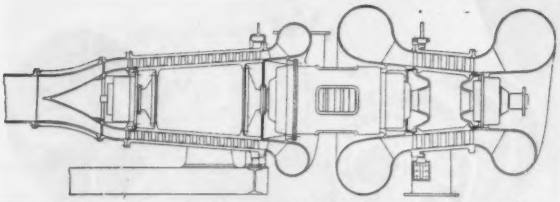

FIG. 5. The Royal Aircraft Establishment—Metropolitan-Vickers B.10 turbocompressor (1939).

The blading used was based on the 1928 tests, and the cambers were much greater than those used in foreign compressors and fans. Testing restarted in October, 1938, and quite promising results were achieved (Fig. 4).

"Anne's" career ended in August, 1940, when it was damaged by blast from an enemy bomb. But soon after "Anne's" career had started, a much more ambitious scheme began to take shape. In March, 1937, H. Constant of the Royal Aircraft Establishment put forward a paper on the internal combustion turbine as a power plant for aircraft. He concluded that such a turbine, designed to drive an airscrew, could be constructed which, on the basis of specific weight and fuel economy, would be superior to, or at least as good as, the best piston engines then current, except when cruising at low heights. He referred to possible developments in materials and compressor design which would make the superiority of the turbine power plant still more marked, and he pointed out the advantage of mechanical simplicity which the turbine engine possessed.

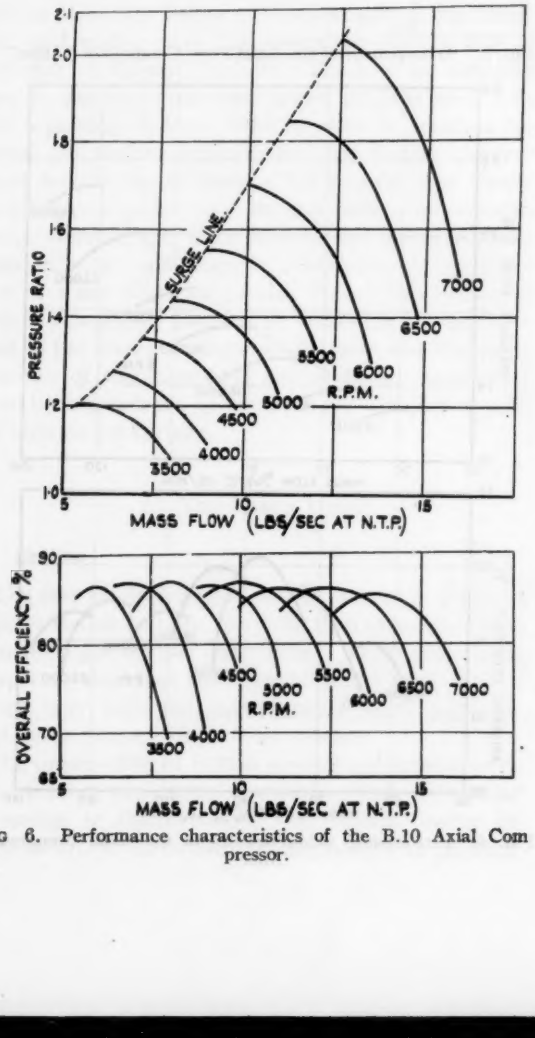

FIG 6. Performance characteristics of the B.10 Axial Compressor.

This paper was discussed in March, 1937, by the Engine Sub-Committee of the Aeronautical Research Committee under the chairmanship of Sir Henry Tizard. At the same meeting, the Sub-Committee considered the Whittle jet propulsion gas turbine scheme. They made appropriate recommendations to the Air Ministry, the results of which are discussed later (§ 2.3) in the case of the Whittle engine. The effect of the recommendations in the case of the turbine-airscrew scheme was that, in April, drawing office work began at the Royal Aircraft Establishment.

The Royal Aircraft Establishment entered into collaboration with the Metropolitan-Vickers Electrical Co. Ltd. The result of their early discussions was a decision to design a plant which consisted of a low-pressure axial compressor driven by a low-pressure turbine in series with a high-pressure axial compressor driven by a high-pressure turbine, and with a separate power turbine driving the airscrew. This arrangement was denoted by the letter "B," and a number of different B layouts was studied. Finally, the decision was reached to go ahead with the high-pressure rotor of the B scheme, of $18^1/_2$-in. maximum diameter over the blading. This plant was designated B.10 and consisted of a nine-stage axial compressor driven by a four-stage turbine (Fig. 5). The compressor first ran in December, 1939, the turbine, in May, 1940, and the complete B.10 unit in October, 1940. The B.10 compressor was, I believe, the first large axial compressor to be tested in England, and its performance was remarkable (Fig. 6). Also early in 1939, Messrs. C. A. Parsons completed and tested an eight-stage axial compressor, "Alice," with blading of R.A.E. design similar to that used for "Anne." Its diameter over the blading was $9^1/_2$ in. "Alice" (Fig. 7) ran more slowly than "Anne" with a smaller pressure rise but higher efficiency (cf. Fig. 9).

In the B scheme to which I have referred, and for which the B.10 high-pressure rotor was constructed and tested, the high- and low-pressure rotors, and the power turbine were on parallel axes and were connected by volutes and crossover ducts. Detailed study of the schemes and experimental evidence of the pressure losses in volutes led to the conclusion that preferably the air and gas should flow straight through the plant without bends, and a compound axial design on this principle was laid before the Metropolitan Vickers Company in October, 1938.

About this time, however, the results from the compressor "Anne" and information from Switzerland indicated that a compression ratio of 5:1 should be ob-

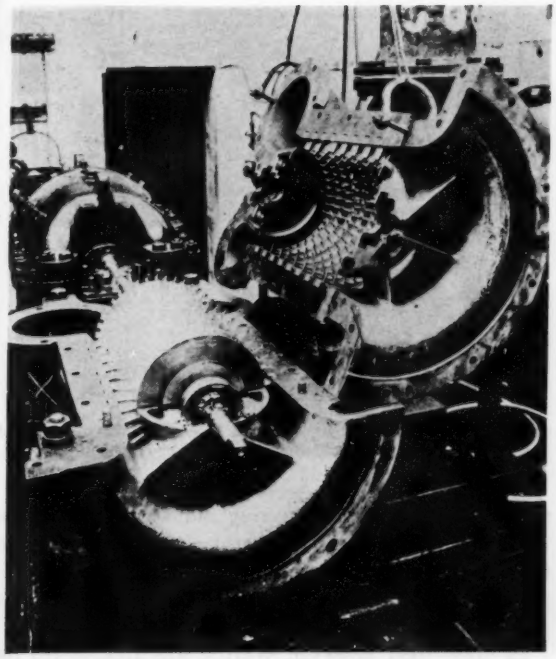

FIG. 7. The Royal Aircraft Establishment—C. A. Parsons Axial Compressor "Alice" (1939).

tainable from a single compressor. This led to a series of schemes in which the air flowed without bends through a single compressor, an annular combustion chamber, the compressor turbine, and a power turbine. Construction finally began in 1940 on a scheme known as D.11 (Fig. 8) designed to give 2,000 s.hp. By this time, however, the potentialities of the gas turbine as applied to jet propulsion were being demonstrated by Whittle, and the clear desirability of applying axial compressors, as well as the centrifugal compressors used by Whittle, to the same end, was manifest. As manufacturing facilities were limited, the result of designing and constructing an axial compressor jet propulsion engine (§ 3.3) seriously slowed down the work on the D.11. Its construction went on slowly through 1940; in 1941 the compressor was tested and found to suffer from Mach number effects. Since then progress has slowed down to a standstill and, as the design is now obsolete, the D.11 will not be completed.

The data obtained from the compressors so far mentioned were augmented through tests of two others. These were known as "Ruth" and E.5. The former was built by the (British) General Electric Co. (Fraser & Chalmers Works) to their mechanical design and the R.A.E. aerodynamic design. It had twice the airflow of "Alice" or "Anne," gave the same pressure ratio as "Anne," but with only six instead of eight stages. E.5 was an axial flow supercharger designed for a compression ratio of 3.7. It was intended to run at full speed only at altitude, and the gas bending blade stresses were high for sea-level operation. The last

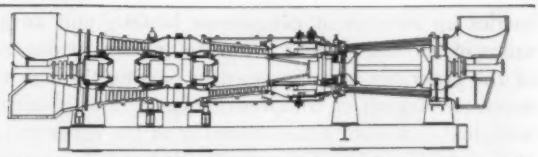

FIG. 8. The R.A.E.—Metropolitan-Vickers Gas Turbine D.11.

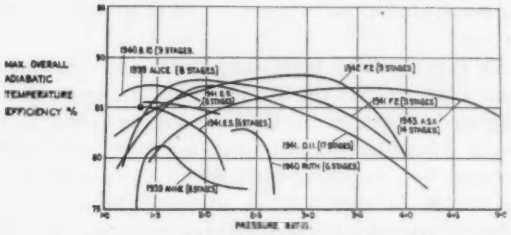
MAX. OVERALL ADIABATIC TEMPERATURE EFFICIENCY %
1939 ALICE (8 STAGES)
1942 FE (9 STAGES)
1941 F.E (9 STAGES)
1941 E.5 (8 STAGES)
1945 ASX (14 STAGES)
1940 RUTH (5 STAGES)
1939 ANNE (8 STAGES)

FIG. 9. Some Axial Compressor Efficiencies.

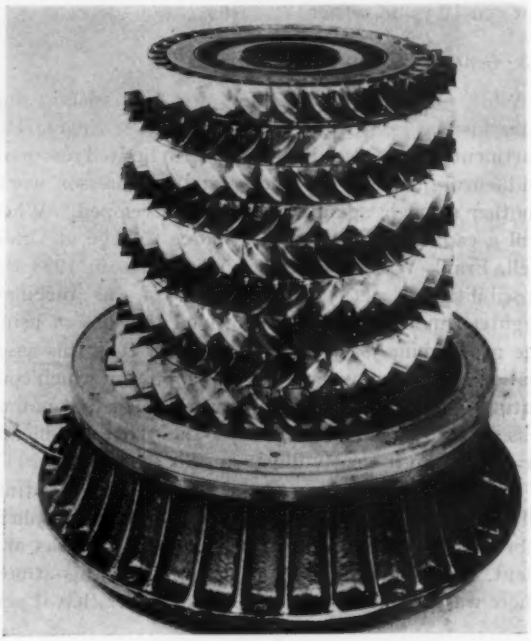

FIG. 10. Griffith contraflow contrarotation turbocompressor proposed in 1929, designed by Royal Aircraft Establishment in 1938, built by Armstrong-Siddeley in 1939, and tested at Royal Aircraft Establishment in 1940.

two rows of blades failed early in the tests at three quarters of full speed, and tests of the remaining six stages were only possible at low speeds. The test results were, however, comparable with those from "Ruth."

The early axial compressor test data have been summarized (Fig. 9).

While these steps forward with axial compressors were being taken, some progress was made with the 1929 Griffith's proposal. In the kind of engine proposed by Griffith, the gas-generating part consists essentially of a number of coaxial wheels, adjacent wheels rotating in opposite directions. Each wheel

carries an annulus of compressor blading and an annulus of turbine blading. The air inhaled at one end of the generator is compressed in passing through the successive annuli of compressor blading and is burned with fuel in a combustion chamber at the other end of the generator. From the combustion chamber the gas passes through the successive annuli of turbine blading and issues therefrom to provide a jet or to drive a power turbine.

An experimental turbocompressor, of 11-in. maximum diameter over the blading (Fig. 10), very much as suggested originally by Griffith, was designed at the Royal Aircraft Establishment in 1938, was manufactured by Armstrong-Siddeley in 1939, and ran at the R.A.E. in 1940. While the performance was indifferent, the running of this turbocompressor, so different in character from its contemporaries, was a notable achievement. One cannot help regretting that it did not run 10 years earlier.

2.3 *Centrifugal Compressor Engines*

While Griffith and Constant, first his assistant and later his successor as Head of the R.A.E. Engine Department, were proceeding slowly with limited resources and encouragement on their axial compressor work, another struggle against orthodoxy developed. While still a cadet at the Royal Air Force College at Cranwell, Frank Whittle described in a thesis in 1928 the possibilities of jet propulsion and of gas turbines. Eighteen months later he conceived the idea of using the gas turbine *for* jet propulsion, and it is this association of the gas turbine and jet propulsion which constitutes the chief novelty of his work. His first patent was dated January, 1930, and the most significant drawing from it clearly illustrates his scheme (Fig. 11).

He submitted his plan to the Air Ministry at a time when (cf. § 2.2), one must conclude, an atmosphere more than usually suspicious of novel ideas was ambient. It would be unfair to assume that this atmosphere was peculiar to the Ministry; I infer that it permeated the British engineering industry, for Whittle was quite unsuccessful in arousing the interest of the firms he approached. The prevailing industrial depression provides the most charitable reason for this lack of enterprise.

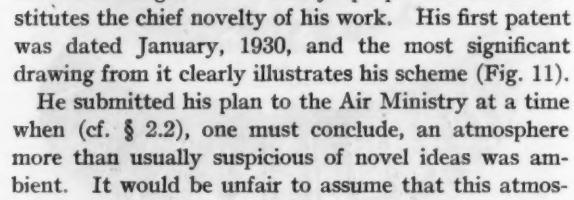

FIG. 11. Scheme drawing from the Whittle British Patent of January, 1930.

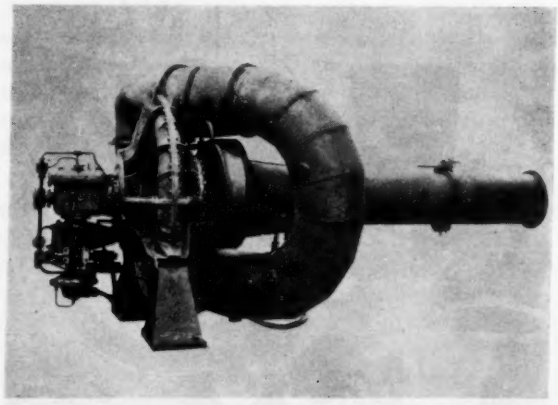

FIG. 12. The first Whittle engine (1937), made to the order of Power Jets Limited by the British Thomson-Houston Co. Ltd.

In May, 1935, however, Whittle was approached by two gentlemen, both invalided out of the Royal Air Force, R. D. Williams and J. C. B. Tinling, the former of whom had been at Cranwell with Whittle. Whittle agreed to work with them and Williams and Tinling finally succeeded in interesting a firm of investment bankers. The bankers, O. T. Falk and Partners, obtained an independent assessment of the Whittle scheme from M. L. Bramson, and on the strength of his favorable report Power Jets Limited was formed in March, 1936.

While the negotiations attendant upon the formation of the company were in progress, the bankers had placed an order with the British Thomson-Houston company for a set of engine drawings in accordance with Whittle's requirements, and in June, 1936, the new company placed the order for manufacturing the engine, except for certain combustion components, instruments, and accessories, with the British Thomson-Houston company.

This engine (Fig. 12) was a simple jet propulsion gas turbine with a single-stage centrifugal compressor, designed to have a compression ratio of 4:1, driven by a single-stage turbine. There was a single combustion chamber between the compressor and the turbine.

This engine was a remarkable venture. Apart from the high compression, the combustion intensity necessary for success was beyond previous experience, and the obviously desirable preliminary stages of separate testing of the main components—turbine, compressor, and combustion system—were too costly in time and money to be possible for the modest resources of Power Jets Limited.

To Whittle, the most difficult part of the design was the combustion system, and he enlisted the aid of

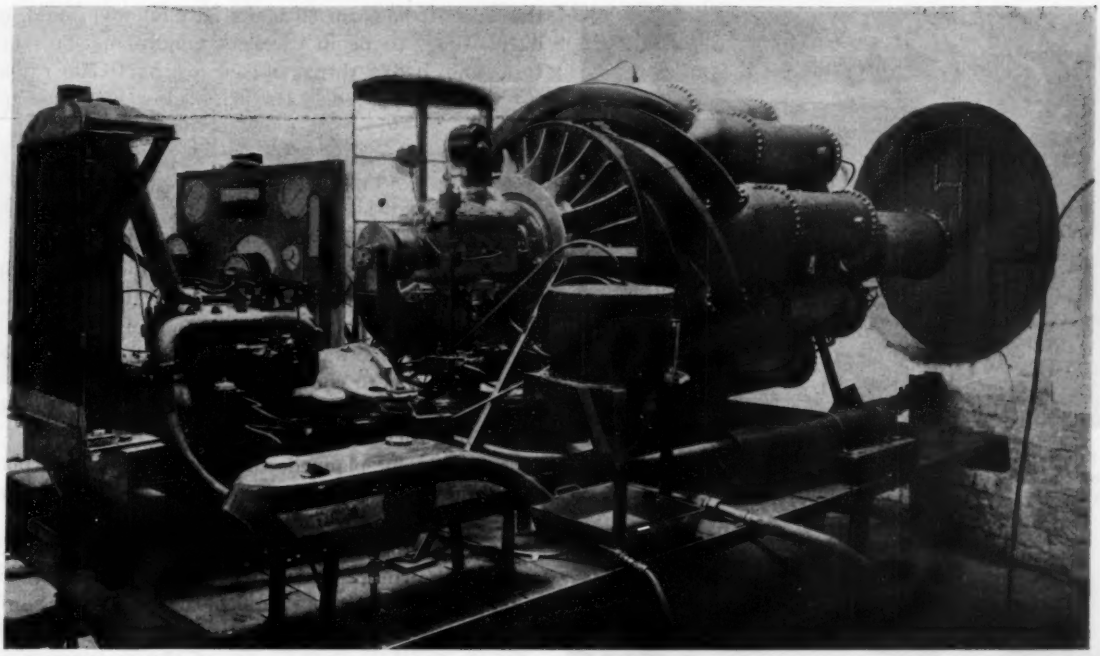

FIG. 13. The first Whittle engine after its second reconstruction (1938).

A. B. S. Laidlaw of Laidlaw, Drew & Co. While the rest of the engine was being designed and made, combustion experiments were done on B.T.H. premises with apparatus supplied by Laidlaw, Drew & Co., and, when sufficient information was deemed to have been collected from these experiments, the same company was given the contract for the design and manufacture of the combustion chamber.

The tests on the complete engine began on April 12, 1937, and continued with intervals until August 23. They showed that the compressor performance was below expectations and that the combustion problem was only partly solved. In consequence, a major reconstruction was put in hand by the British Thomson-Houston company for Power Jets Limited.

Meanwhile, as I have already noted (§ 2.1), the Aeronautical Research Committee had discussed gas turbines afresh, and their recommendations of May, 1937, no doubt influenced the Air Ministry to take more interest in Whittle's efforts. The Ministry decided to place contracts with Power Jets Limited for a report on the tests of April to August, 1937, and for further experiments with the engine.

While the reconstruction of the engine was in progress, further combustion experiments were done, the lessons from which were embodied in the unit. It ran on April 16, 1938, and testing continued intermittently until a turbine blade failure caused considerable damage on May 6. The combustion was still clearly unsatisfactory.

Further combustion testing and another reconstruction, this time under Governmental contract, were put in hand, and the third version (Fig. 13) of the unit was on the test bed at the end of October, 1938. In the subsequent trials, better fortune attended this gallant precursor, and it was worked until February, 1941, when, full of hours and honor, it came to its end through a turbine failure.

Long before this, however, in the summer of 1939, the Air Ministry had concluded that on the evidence by then accumulated by Whittle, he had, in essence, a practical aircraft engine. They therefore commissioned Power Jets Limited to supply an engine for flight test. A contract was also given to the Gloster Aircraft Co. Ltd., for the design and construction of an experimental airplane. This aircraft, which has claims to renown like those of "Puffing Billy," "Rocket," "Britannia," and "Turbinia," will go down to posterity by the mere number of its specification—E.28/39. But still, the Wright airplane did not, to the best of my knowledge, have even a number.

The flight engine, known as W.1, was designed by Whittle and the engineers of the British Thomson-Houston company, to whom the drawing and manufacture were again subcontracted by Power Jets Limited.

From unairworthy parts and spares of the W.1 engine a bench engine called W.1X was assembled and run by Power Jets Limited during 1940, and when the W.1 engine was delivered to Power Jets, the company incorporated in it the results of the experience gained with the bench engine. Then, after preliminary bench tests, the W.1 engine (Fig. 14) was put through a 25-

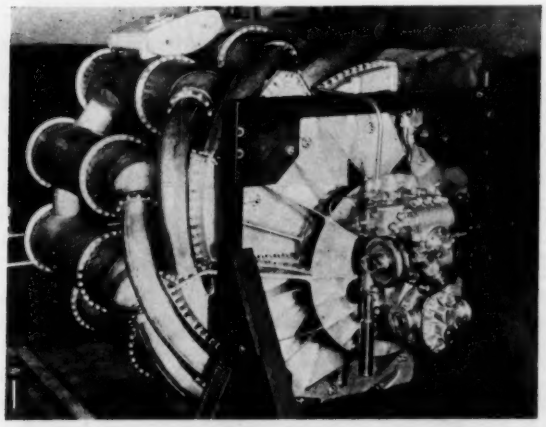

FIG. 14. The W.1 engine built by the British Thomson-Houston Co. Ltd. for Power Jets Limited and used in the first flights (1941).

FIG. 15. The E.28/39 aircraft built by the Gloster Aircraft Co. Ltd., Chief Designer W. G. Carter, which flew on May 15, 1941.

hour bench test to clear it for flight test. The clearance permitted approximately 10 hours flying with the engine.

2.4 The First Flights

In the meantime, the E.28/39 (Fig. 15) had been completed and during April, 1941, was taxied with the W.1X engine installed. In the course of these taxiing trials at Hucclecote, the Gloster Company's aerodrome, the aircraft left the ground—no doubt through sheer inadvertence!

Flight Lieutenant P. E. G. Sayer began the trials with the flight engine, W.1, on May 14, 1941. In the evening of that day he did taxiing trials, and on May 15 he did the first real flight, being air-borne for 17 min. and doing speed runs at 2,500 and 4,000 ft.

The flight program was completed in a fortnight, the 17th and last flight of this first series being completed on May 28. The total flying time was 10 hours 28 min. and had been completed without incident. The aircraft, which had in most of its 17 flights taken off at 3,691 lbs. all-up weight, with 81 Imperial gallons of paraffin fuel, performed admirably. The engine which, including its 25-hour bench test and ground running in the aircraft, had run 39 hours 57 min. was found, on dismantling, to be in excellent condition. This historic unit had a thrust of 850 lbs. at 16,500 r.p.m., its maximum speed. Its installed weight was 623 lbs.

3. The Gas Turbine Engines

3.1 The Power Jets Engines

As the design work on the E.28/39 tapered off during 1940, Mr. Carter and his colleagues at the Gloster Aircraft Company began to give their attention to its successor. This was designated F.9/40. Whereas E.28/39 had been conceived as an airplane for experiment and research, F.9/40 was designed as a fighter for the Royal Air Force.

As originally planned, the engine for this airplane was the W.2, which the Power Jets company was authorized in 1940 to design and construct.

The drawings of the W.2 were handed over to the Rover company, who also constructed it, but with certain changes in mechanical design, under direct contract to the Ministry of Aircraft Production. In this engine, the design efficiencies of the compressor and turbine were not achieved and, as a result, it did not equal expectation and was subject to surging.

Power Jets, by a series of modifications, including complete changes in blower casing and diffuser design, brought their W.2 engine, which had been built for them by the British Thomson-Houston company, to a relatively satisfactory condition, known as W.2.Mk.IV, in which it differed only slightly from their then latest design, W.2B (Fig. 16).

W.2B must rank among the classics. It was the design from which the F.9/40, its production version the Meteor, and the Bell P-59 derived their power plants. Its development has been along two main lines. First,

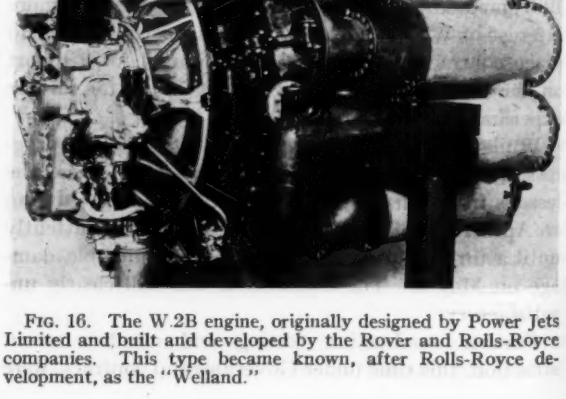

FIG. 16. The W.2B engine, originally designed by Power Jets Limited and built and developed by the Rover and Rolls-Royce companies. This type became known, after Rolls-Royce development, as the "Welland."

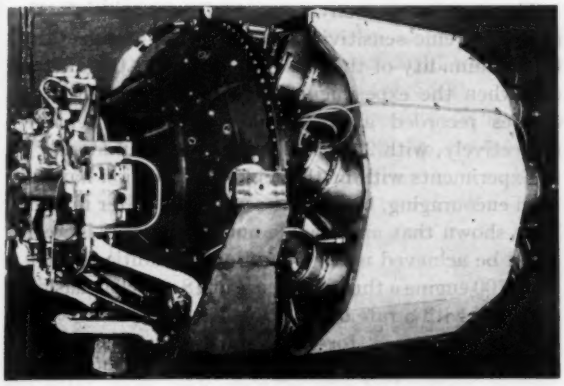

FIG. 17. Power Jets W2/700 engine (shown with heat shield before installation in E.28/39) with original blower casing and diffuser system.

the development by Power Jets to what appeared to be the limit with the combustion system as originally conceived. Secondly, the development by the Rover Company and, latterly, Rolls-Royce, using a "straight-through" combustion system.

Restricting consideration for the moment to the Power Jets line of development, the most important aspect has been the step-by-step increase of throughput by successive lengthenings of the turbine blades. Whereas in the W.2B engines the turbine blades were 2.455 in. long, in the W.2/500 design, the first example of which ran on September 13, 1942, the blades were 2.73 in. long. The top speed performance of this engine is worth recording—

r.p.m.	16,750
Thrust (lb.)	1,755
Jet temperature (°C.)	606
Fuel consumption (lb./lb. thrust/hour)	1.13

The next step was to make the blade height at the trailing edge 0.3 in. greater (i.e., 3.03 in. overall) leaving the height at the leading edge the same. This change corresponded to an increase in throughput of air from 34.73 lb./sec. to 40.33 lb./sec. The performance was improved, and the further step of making the blade height 3.03 in. at inlet and outlet made no appreciable difference. A typical set of performance figures at this stage was

r.p.m.	16,750
Thrust (lb.)	1,850
Jet temperature (°C.)	620
Fuel consumption (lb./lb. thrust/hour)	1.12

A complete change of diffuser design was next successfully tested; the air, instead of being led in smoothly varying channels from the impeller to the combustion chambers (Fig. 17) was swung out in the plane of the impeller disc and then turned through 90° before being led in fore-and-aft channels to the combustion chambers (Figs. 18 and 19). This produced a worth-while improvement—

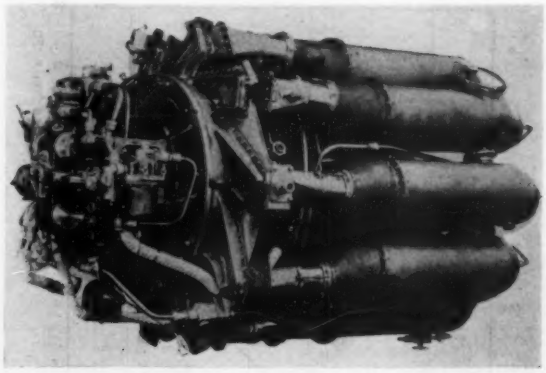

FIG. 18. Power Jets W.2/700 engine in final form.

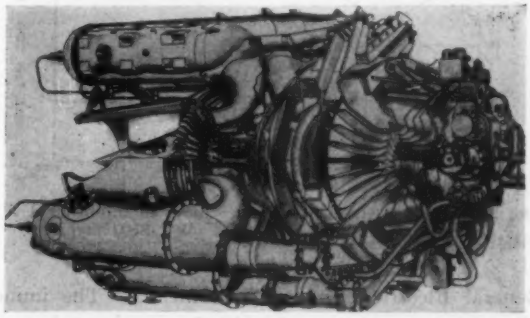

FIG. 19. Cutaway view of Power Jets W.2/700 engine with second blower casing and improved diffuser system.

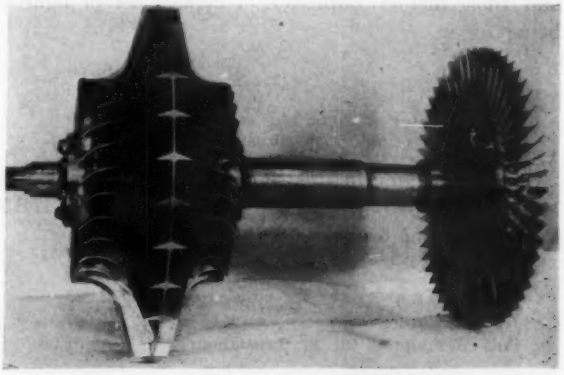

FIG. 20. Rotor of the Power Jets W.2/700 engine.

	Old Diffuser	New Diffuser
r.p.m.	16,750	16,750
Thrust (lb.)	1,850	2,040
Jet temperature (°C.)	620	597
Fuel consumption (lb./lb. thrust/hour)	1.12	1.07
Delivery pressure (lb./in.²)	43.5	45.0

It was in these engines which, both with old and new diffusers (Fig. 20), were known as W.2/700, that a series of impeller failures occurred. The current design of impeller was therefore replaced by a safe design based upon that used in engines of the same kind by the

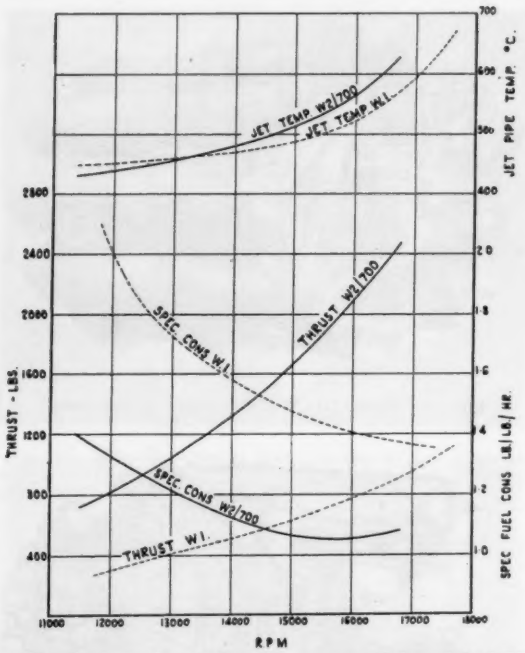

FIG. 21. Comparative performance curves of the original W.1 engine and a W.2/700 batch IV (W.2/850) engine.

General Electric Company of America. The immediate result was a drop of 150 lbs. in thrust but, as work proceeded, not only were there no more impeller failures, but the performance was regained and then further improved, a typical set of figures being

r.p.m.	16,750
Thrust (lb.)	2,130
Jet temperature (°C.)	646
Fuel consumption (lb./lb. thrust/hour)	1.077

The last engine in this series had the leading edge blade height 3.63 in. The throughput was 47.15 lb./sec. and the top-speed performance—

r.p.m.	16,750
Thrust (lb.)	2,485
Jet temperature (°C.)	647
Fuel consumption (lb./lb. thrust/hour)	1.05

This engine fits into the same space as W.1. Its thrust (Fig. 21) is nearly three times as great, while its weight has increased only 50 per cent.

Power Jets engines have been the vehicles for certain experiments on thrust boosting which may appropriately be mentioned at this stage. The earliest experiments were with ammonia injection, and typical results are

Engine	$\frac{\text{Weight of } NH_3}{\text{Weight of Air}} \times 100$ (Per Cent)	Relative Humidity of Ambient Air (Per Cent)	Fall in Temperature of Jet (°C.)	Increase in Thrust (Per Cent)
W.1 Mk.III	6.15	85	115	28
W.2B	4.4	81	72	22.4

The most unattractive aspect of ammonia injection is the extreme sensitivity of the thrust augmentation to the humidity of the atmosphere. Had the air been dry when the experiments were made, the thrust increases recorded above would have been achieved, respectively, with 2.8 and 2.3 per cent of ammonia.

Experiments with methyl chloride injection have not been encouraging, but experiments with water injection have shown that appreciable and dependable increases could be achieved under operational conditions. On a W.2/700 engine a thrust increase of 18 per cent has been obtained with a rate of injection of 102 lb./min.

Another device for temporarily augmenting the thrust of a jet propulsion gas turbine, which has also been tested on Rolls-Royce Whittle-type engines, has been termed "reheating." The jet of gas issuing from the turbine contains great quantities of unburnt oxygen, so that it can be "reheated" by burning additional fuel in it. This is, obviously, an inefficient process thermodynamically but is nevertheless a worth-while expedient for obtaining short bursts of high thrust such as may be required for take-off or combat. Typical results obtained on the bench with a W.2B/23 engine (cf. § 3.4) are

Measured Increase in Fuel Consumption (Per Cent)	Measured Increase in Static Thrust at Sea Level (Per Cent)	Estimated Increase in Thrust at 500 M.P.H. at Sea Level (Per Cent)
30	5	9
70	10.8	17
100	16	27.5

Improvements in performance in flight trials with a different engine are consistent generally with the order of increase of thrust given in the last column.

These schemes for thrust boosting demand, for thorough exploitation, the use of variable area nozzles for the jet. If on a particular engine the exit nozzle area is reduced, both thrust and jet temperature increase. Injection of ammonia or water reduces the combustion temperature, and, consequently, the exit nozzle area may be reduced until the jet temperature is raised to the acceptable limit, or some other limiting factor such as compressor surging operates. If nozzle area is left unchanged, reheating will cause an increase in combustion temperature, and to use it to best advantage the nozzle area should be increased when the reheat is turned on.

A number of different variable nozzle schemes has been tried, the work having been begun by the R.A.E. turbine staff before their amalgamation with Power Jets (Research and Development) Limited. So far the best of them is one employing a caliper device (Fig. 22). As the calipers are closed, the jet issues through a sharp-edged orifice, but no consequent ill effect on engine performance has been observed. From the aerodynamic viewpoint, the flat faces of the calipers

appear likely to produce drag in flight, but any such consequence is not important enough to have been detected yet in the relatively slow speed flight trials so far done in a Wellington flying test bed (cf. § 6.3).

Power Jets are one of the two companies to have experimented with the ducted fan (Fig. 23), so far only on the test bed.

3.2 The British Thomson-Houston Engines

The major contribution so far made by the British Thomson-Houston company has already been sketched in the references I have made to the collaboration between that company and the Power Jets company in the early days. The first aircraft gas turbine engines were made in their Rugby factory, and they have in consequence an honored place in gas turbine history. If I may be permitted to speculate, I would prophesy that their contributions in the future will not be in the aircraft field but in applications to heavier engineering. They have, however, remained in close touch, through the collaboration system that has been developed, with all the work described in this paper and have conducted a series of experiments with their own version of the W.2B design. This design (Fig. 24) has a distinctive appearance and engines of the kind are still running on experimental investigations. The maximum static sea-level performance is 1800-lb. thrust at 660°C. jet temperature and 1.17 lb./lb. thrust/hour specific fuel consumption.

3.3 The Metropolitan-Vickers Engines

I have already indicated (§ 2.2) that the axial compressor work in which the Royal Aircraft Establishment and the Metropolitan-Vickers Electrical Co. Ltd. were associated, in which the objective was the development of a propeller gas turbine, was interrupted by the clear necessity, demonstrated by the Power Jets achievements, of first achieving an axial compressor jet propulsion gas turbine. Design work on an engine of this kind was begun by the Metropolitan-Vickers company in July, 1940, on the basis of preliminary design work done under Mr. Constant at the Royal Aircraft Establishment. This engine was designated F.2 (Fig. 25). It had a nine-stage axial flow compressor, a single annular combustion chamber, and a two-stage turbine, all on the same axis (Fig. 26).

Bench testing of the first engine to this design began in December, 1941, and after modifications the engine passed a 25-hour special category test in November, 1942.

A pair of these early engines was installed in a specially modified Gloster F.9/40 aircraft (Fig. 27), and the first British flight of an axial compressor type jet propulsion gas turbine aircraft took place at Farnborough, Hampshire, on November 13, 1943.

Another of these engines was tested in a flying test bed (cf. § 6.3). The first flight was made on June 29,

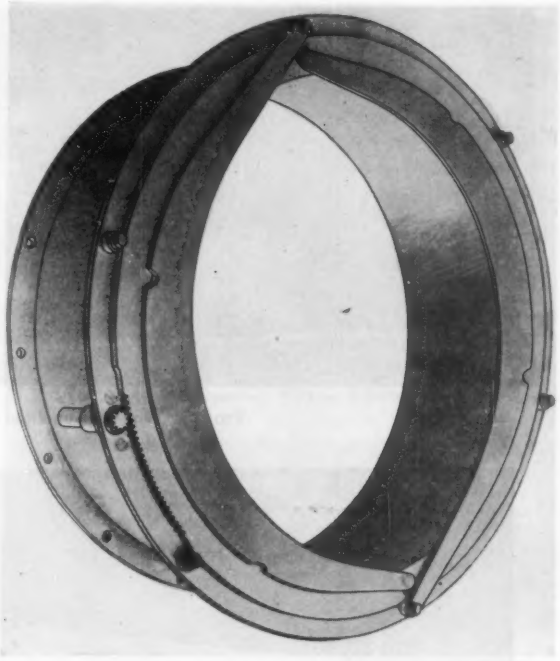

FIG. 22. Arrangement of "caliper" type of variable jet nozzle.

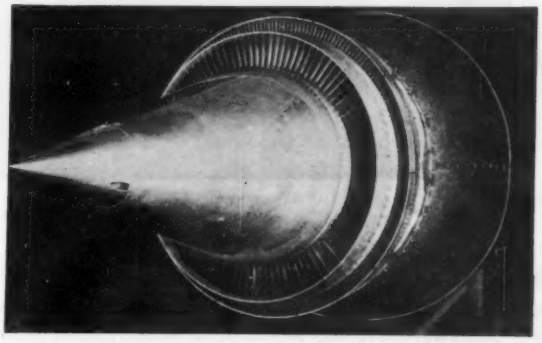

FIG. 23. Power Jets experimental ducted fan for W.2/700 engine.

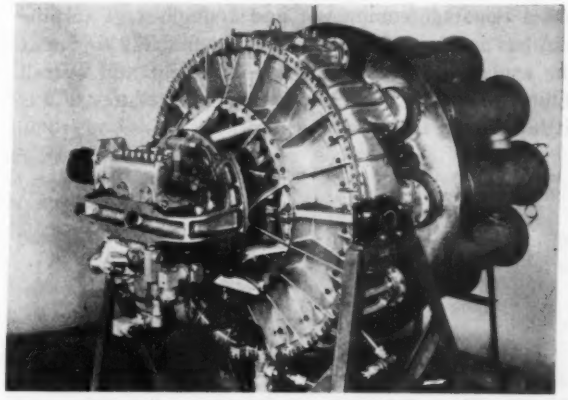

FIG. 24. British Thomson-Houston Whittle type engine.

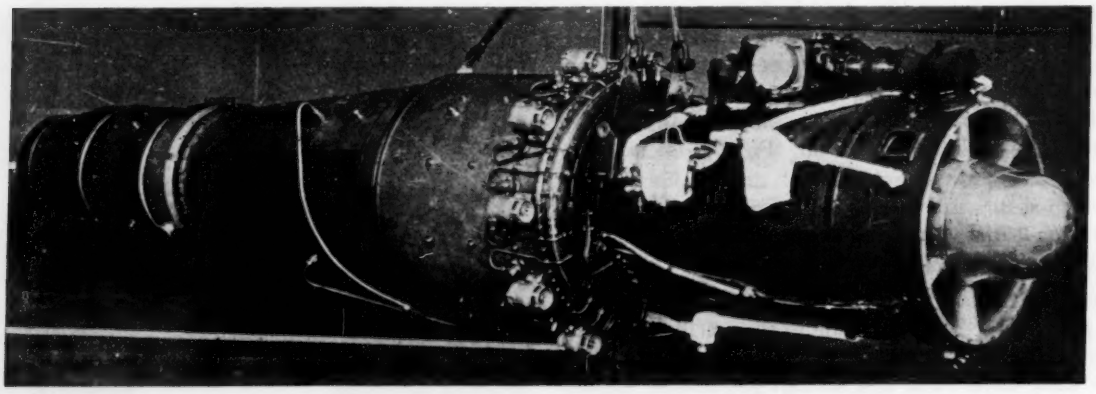

FIG. 25. The Metropolitan-Vickers Company F.2 engine.

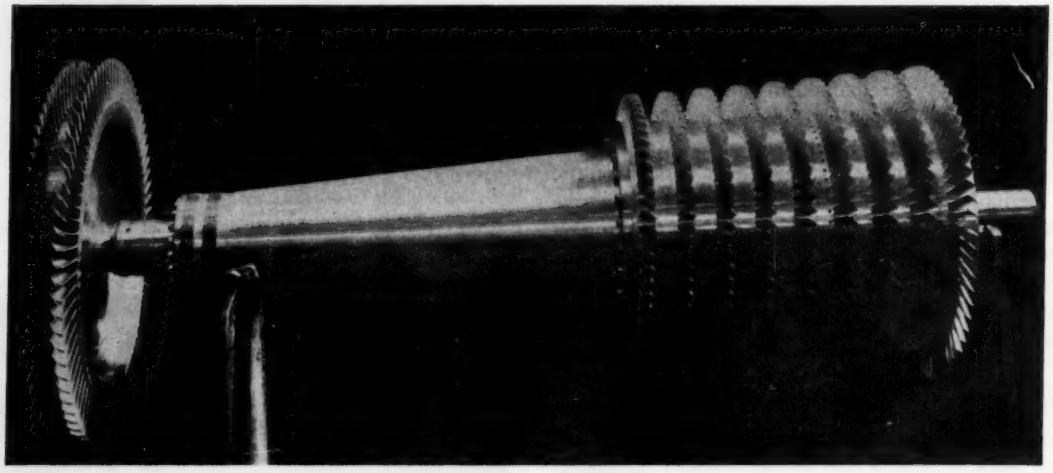

FIG. 26. Rotor of the Metropolitan-Vickers F.2 engine.

1943. The engine under test ran 210 hours in completing its program of work, 90 hours of which were in the air. Afterwards, it was found on strip examination to be in good condition.

The Metropolitan-Vickers company has continued to develop the F.2-type engine. In its latest form it has a ten-stage compressor and a single-stage turbine and has a thrust of twice that of the first F.2 engine at the expense of small increases in weight and overall dimensions (Fig. 28). In appearance, the latest F.2 is little different from its progenitor.

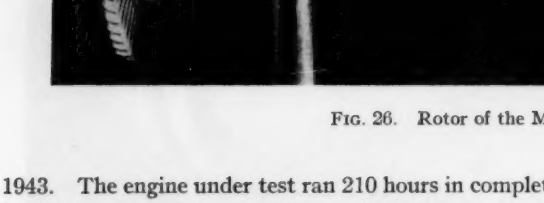

FIG. 27. Gloster F.9/40 aircraft with Metropolitan-Vickers F.2 engines (1943).

It is to this company also that the credit for designing and constructing the first ducted fan augmenter (Fig. 29) belongs (cf. § 7.4). The combination of F.2 engine and ducted fan was designated F.3 and bench testing on it began in August, 1943. Most promising results have been obtained.

3.4 The Rover Engines

In the early spring of 1940 the decision was taken by the Air Ministry to bring the Rover company into the jet propulsion engine field to manufacture the Whittle W.2 type of engine in quantity, the intention being that the Power Jets company would do the research and development while the Rover company did the production and any development necessary to facilitate production.

In April, 1940, Power Jets handed over most of the drawings of the W.2 engine to the Rover company. The next 12 months were spent by the Rover company, under conditions rendered particularly difficult by enemy action, in drawing and making the first W.2

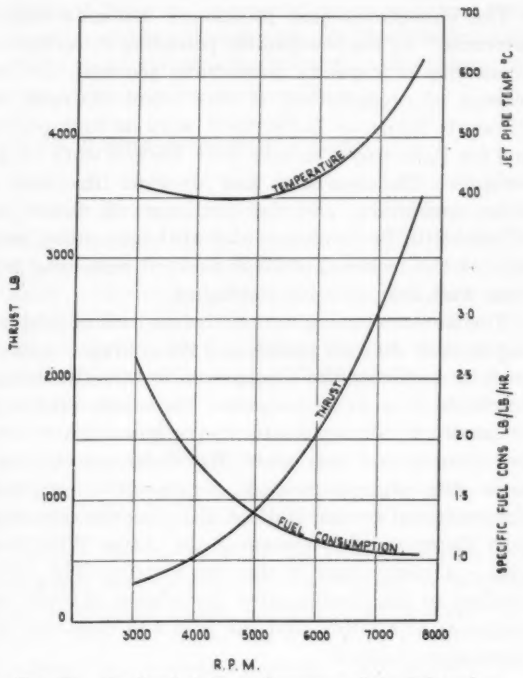

FIG. 28. Sea-level static performance of the F.2 engine.

engine, and, at the same time, in preparation for production. As was to be expected, the engine as made differed in some respects from the Power Jets design, but the differences were mainly mechanical (for example, in the gear box and accessories), the aerodynamic and thermodynamic features remaining unaltered.

The first W.2 engine running was begun at Power Jets in April, 1941. The results were as already mentioned (§ 3.1), disappointing; the engine surged at a speed well below the maximum designed r.p.m. The modified engine, the W.2.Mk.IV, although better than the W.2 was still, however, prone to surging at a speed which prevented attainment of maximum r.p.m.

In parallel with the developments to eliminate surging, arrangements had been going forward at the Rover factory at Barnoldswick to produce W.2B engines for the F.9/40 and Meteor aircraft and, toward the autumn of 1941, the M.A.P. concluded that development would have to be intensified to avoid a holdup in production.

A policy of competitive development was then adopted and the Rover company were encouraged by the Ministry of Aircraft Production to press ahead with its ideas to improve performance and reliability of the engine.

In addition to the need for eliminating surging, there were the needs for increasing the rated performance to the design figure for the Meteor (1,600-lb. static thrust at sea level) and for overcoming the mechanical defects which the process of development is designed to bring to light and cure.

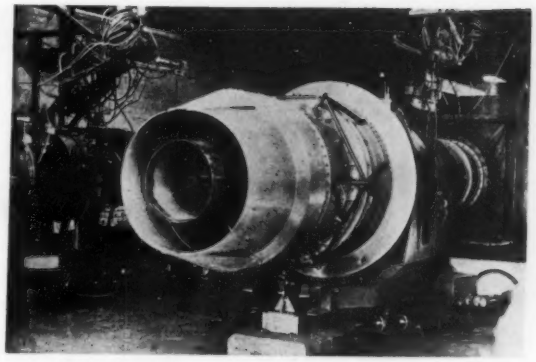

FIG. 29. The Metropolitan-Vickers F.3 thrust augmenter.

FIG. 30. Gloster F.9/40 aircraft with Rover-built W.2B engines (1942).

FIG. 31. Bell XP-59A aircraft with G.E.C. type I (corresponding to Rover-Power Jets W.2B) engines (1942).

Engines were installed in an F.9/40 aircraft in the summer of 1942 and taxiing trials began on July 10 of that year. With the runway lengths available, however, the aircraft (Fig. 30) was not allowed to take off, and sufficient improvement for flight was delayed until June, 1943. Meanwhile the Bell XP-59A, with two G.E.C. Type I engines, corresponding to the Rover-Power Jets W.2B engines, had been flying since October 1, 1942 (Fig. 31).

The Rover Company introduced skewed turbine blades and the B.23 design of blower casing and diffuser system. They also made considerable progress in bringing the engine, designated W.2B.23, effectively to the form, known as the Welland, in which it went into production Meteor aircraft. The final stages of the process were, however, as will be made clear later (§ 3.6) in the hands of the Rolls-Royce company.

Early in 1940, Power Jets had schemed a version of the Whittle engine with a so-called "straight-through" combustion system in place of the return-flow system. Power Jets did not regard this design with favor at the time because of the development period involved; for,

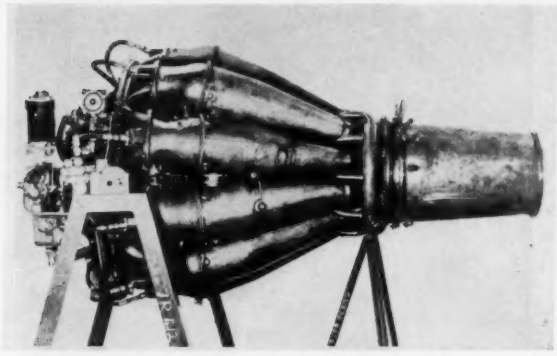

FIG. 32. The Rover W.2B.26 engine (1942).

in addition to the new combustion system, it involved a much longer shaft between compressor and turbine and a third bearing. In the eyes of the Rover company, however, then much concerned with production problems, the scheme had the great merit of easing the manufacture of the sheet-metal combustion system components, and they decided to try it. Their prototype, known as W.2B.26 (Fig. 32) ran in March, 1942. Although this actual design did not go into service, it served as the prototype of the Derwent I engine, developed by Rolls-Royce for the Meteor III aircraft. In fact, the original design of the Derwent I was begun by Rovers at the request of Rolls-Royce during the transition period before that company formally took over the Rover work on April 1, 1943.

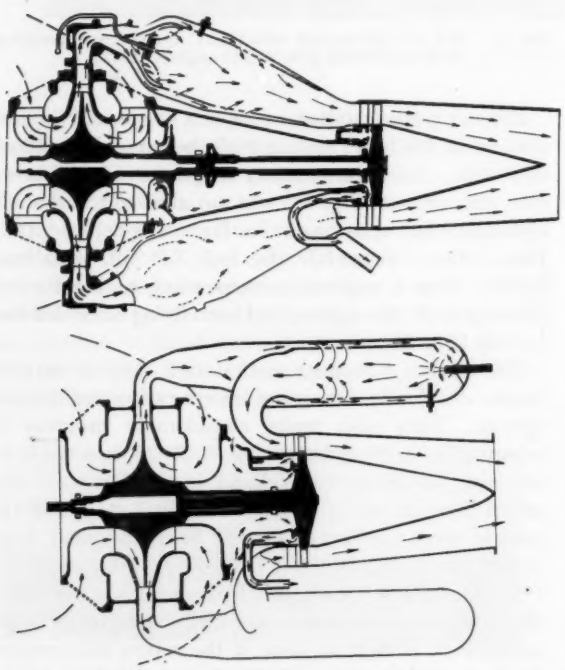

FIG. 33. Comparison of "return-flow" and "straight-through" combustion systems on double-sided impeller engines.

The changeover just mentioned demands further comment. In the few months preceding it, certain reallocations of capacity were being planned, and for reasons of organization, it was found desirable for Rovers to take over Rolls-Royce work on tank engines and for Rolls-Royce to take over Rover's work on gas turbines. The two firms had for some time been in close association, and the exchange of duties was effected with the minimum of disturbance, many members of the technical staffs concerned remaining with their work and changing employers.

The Rover company were in the gas turbine field during its most difficult period, and the courageous step it took in initiating the changeover to straight-through combustion is in consequence the more creditable. Naturally, the change occasioned technical controversy, but there is now little doubt that under current conditions the straight-through arrangement, originally demonstrated on the Halford H.1, has the advantage over the return flow system of the classic Whittle engine. A comparison of the two systems (Fig. 33) as applied to the double-entry compressor Whittle-type engine is of interest, and the pros and cons are summarized in Table 1.

TABLE 1

	Return Flow Combustion System			Straight-Through Combustion System
1.		+	−	Requires a longer shaft, involving a third bearing and a flexible coupling.
2.	Permits automatic compensation of expansion without special joints.	+	−	Requires expansion joints to allow differential expansion.
3.	Is such that the turbine stator and rotor blades are shielded from direct high-temperature radiation from the flame.	+	−	Subjects turbine stator and rotor blades to direct high temperature radiation from the flame with possibly higher metal temperatures.
4.	Involves two 180° bends in the path of the gas.	−	+	Involves no bends and so has the lower pressure loss.
5.		−	+	Allows the greater cross-sectional area for combustion within a given overall diameter.
6.		−	+	Is the simpler manufacturing proposition.
7.	Allows easier location and inspection of burners and easier assembly and removal of combustion chambers.	+	−	
8.		−	+	Should provide the more even air distribution.

If the points made were of equal weight, the merits of the two systems would be evenly balanced. Item 5

in Table 1, however, represents a preponderating design advantage in the larger engines, and the disadvantages noted against the straight-through system have, with development, shrunk in importance.

3.5 The Halford-de Havilland Engines

Frank B. Halford, with a series of notable piston engines to his credit, was brought into the gas turbine field in January, 1941, at the instance of Sir Henry Tizard, who was then at the Ministry of Aircraft Production. The organization bearing Halford's name worked on the gas turbine problem, as on many other problems, in intimate association with the de Havilland Aircraft Company Limited. This intimate association became, without any change in personnel, in March, 1944, an identity, the de Havilland Engine Company Limited of which the Chairman and Technical Director is Halford. These facts are noted in explanation of the hyphenated title of this section of the paper.

Halford, on the basis of his long experience with centrifugal blowers and with a knowledge of Whittle's progress with his engine, decided to use a single-sided impeller (Fig. 34) in contrast to Whittle's double-sided impeller, and a "straight-through" combustion (§ 3.4) system in contrast to the return flow system favored by Whittle. With a single-sided impeller, straight-through combustion does not (cf. § 3.4) demand a third bearing; and the whole conception of the H.1 (Figs. 35 and 36), later to be known as the "Goblin," had an elegant simplicity.

Design work started in April, 1941, and testing of the first H.1 began in April, 1942. A pair of the engines was installed in an F.9/40 aircraft (Fig. 37) which flew first on March 5, 1943. This was the first F.9/40 aircraft to fly as, although W2B engines had been installed in another F.9/40 in the previous July, a series of difficulties had prevented it flying until June, 1943 (§ 3.4). At the time of the first F.9/40 flights, the H.1 was rated at a maximum static sea-level thrust of 2,000 lb. Later, the de Havilland company received the first technical certificate, dated February 27, 1945, issued for a full service type test on the H.1 or Goblin engine, at 2,700-lb. maximum static sea-level thrust. Now, in the Vampire (Fig. 38) which first flew in September, 1943, it is rated for production at 3,000 lb. (Fig. 39). A considerable amount of bench and flight testing has been done at 3,400–3,500-lb. thrust.

The H.1 was the engine of the prototype Lockheed XP-80, which flew first in January, 1944. It has also

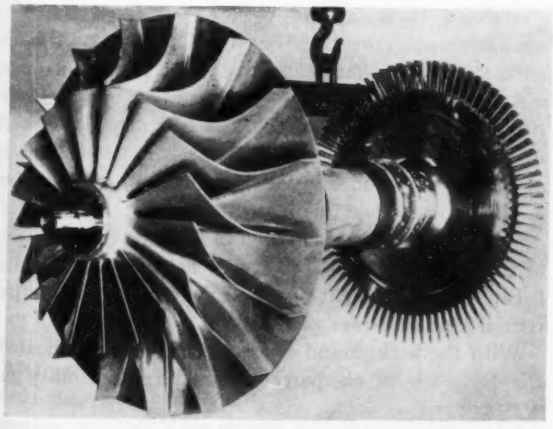

FIG. 34. Rotor of the Halford H.1 engine.

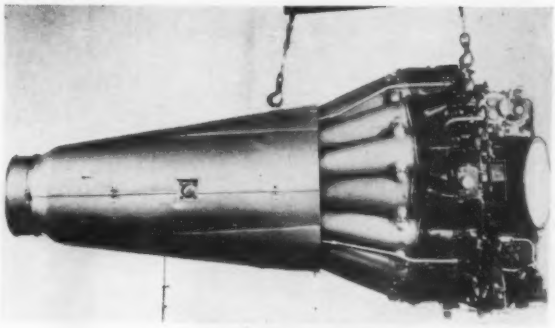

FIG. 35. The Halford H.1 engine.

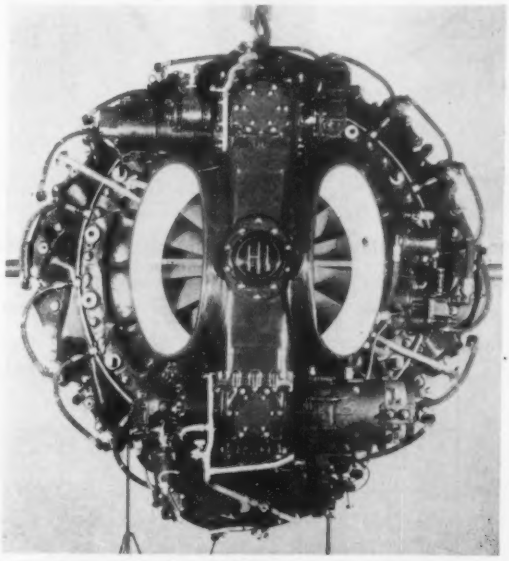

FIG. 36. Front view of the Halford H.1 engine showing intakes.

FIG. 37. Gloster F.9/40 aircraft with Halford H.1 engines (1943).

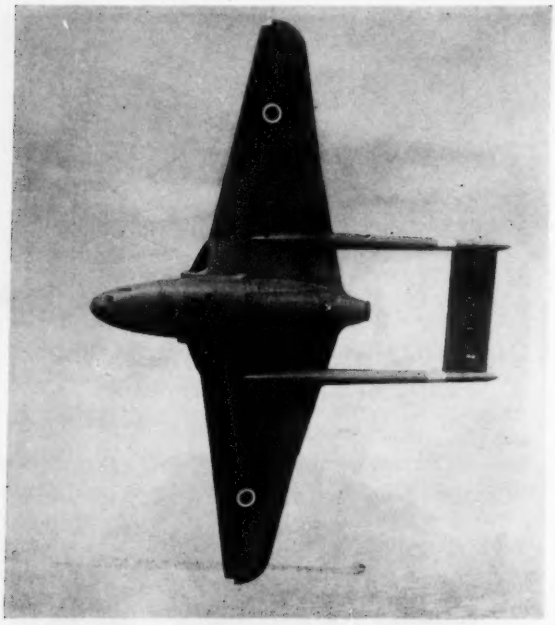

FIG. 38. De Havilland Vampire aircraft with de Havilland-Halford Goblin engine.

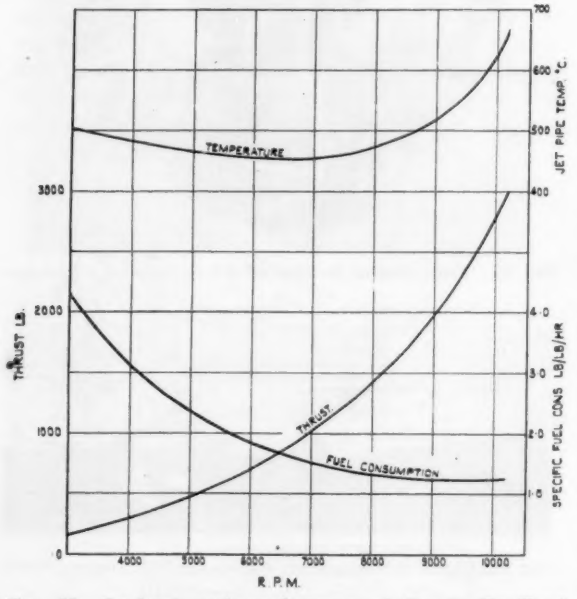

FIG. 39. Sea-level static performance of the de Havilland Goblin II production engine.

been flown in the U.S.A. in a converted Grumman XTBF3-1 flying test bed and the Curtiss XF-15C Navy fighter. Some of the later engines for these aircraft were made by the Allis-Chalmers company under arrangements sponsored by the British and American governments.

There is, not unnaturally, some controversy about the respective merits of single- and double-sided centrifugal impeller engines. The double-sided impeller is naturally of considerably smaller diameter for the same throughput than the single-sided impeller. The exponents of the former consequently claim that their engines are of smaller frontal area for a given thrust than single-sided impeller engines. Against this several counter claims are urged. For example, there is the tendency for combustion chamber volume to control engine diameter; there is the lower rotational speed of the larger impeller, which relieves turbine centrifugal stresses; there is in the double-sided impeller engine the need to have a nacelle diameter somewhat greater than engine diameter in order to allow the intake air to reach the rear intakes. The methods of leading the intake air to the compressors are different in the two kinds of engine: in the case of the single-sided impeller engine the air can conveniently be ducted directly into the impeller eye; in the double-sided impeller engine some form of plenum chamber is required. It is clear that the advantage of small impeller diameter is balanced to a more or less degree by other influences according to the demands and tastes of the aircraft designer, and argument dissociated from particular aircraft designs is apt to be somewhat unprofitable (cf. § 7.3).

3.6 The Rolls-Royce Engines

I think it can be said that the Rolls-Royce interest in gas turbines dates at least from the time when Dr. A. A. Griffith left the Royal Aircraft Establishment in the summer of 1939 to join the company. Since then the Rolls-Royce company have made a number of experiments on the contraflow—contrarotating class of engine first proposed by Dr. Griffith in 1929, but there still remain a number of mechanical and thermodynamic problems to be solved before this class of engine fulfills its great promise.

Exactly when the Rolls-Royce interest in the very different Whittle-type engine began is difficult to decide. They had, necessarily, been for many years interested in the development of centrifugal blowers for their reciprocating engines when the first flight of the E.28/39 took place in May, 1941. This major event stimulated their interest, which, sustained by their membership of the Gas Turbine Collaboration Committee (§ 4.1), culminated early in 1942 in an arrangement by which Power Jets subcontracted to Rolls-Royce Limited an order from the Ministry of Aircraft Production for an engine known as WR.1.* This was an engine based on the W.2B, incorporating a number of Rolls-Royce ideas on blower, turbine, and mechanical design. Its development was in fact the development of the Rolls-Royce skill in a new field, and the engine became an instrument of experiment rather than a potential aircraft power plant.

With the background of experience I have indicated, the Rolls-Royce company assumed responsibility for

* The initials were those of Whittle and Rolls-Royce and symbolized the close technical liaison.

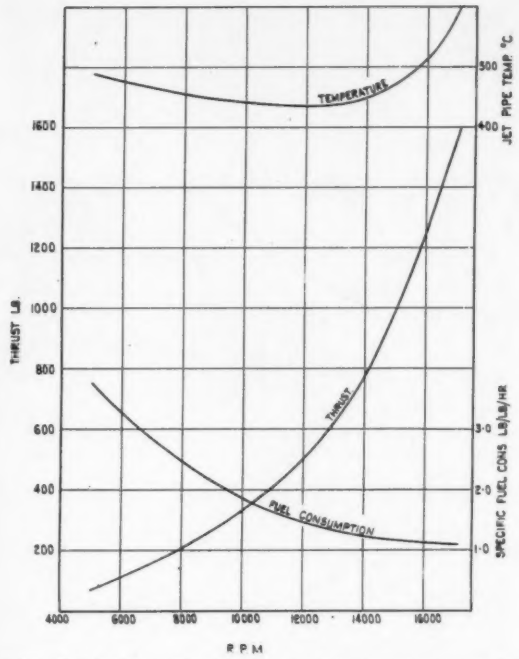

FIG. 40. Sea-level static performance of the "Welland" production engine.

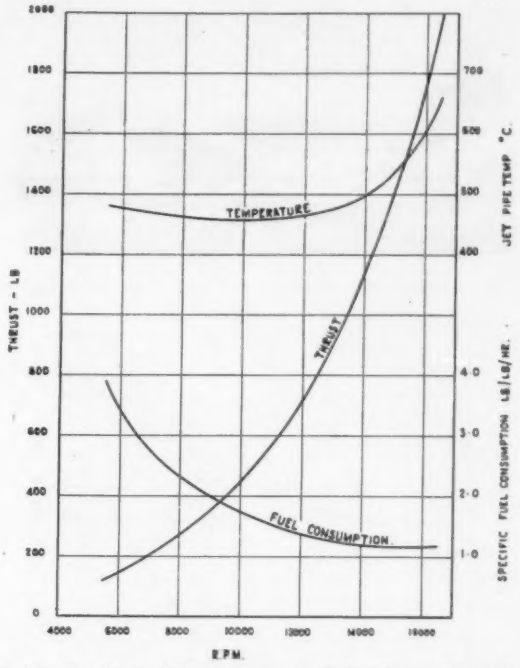

FIG. 43. Sea-level static performance of the Derwent I production engine.

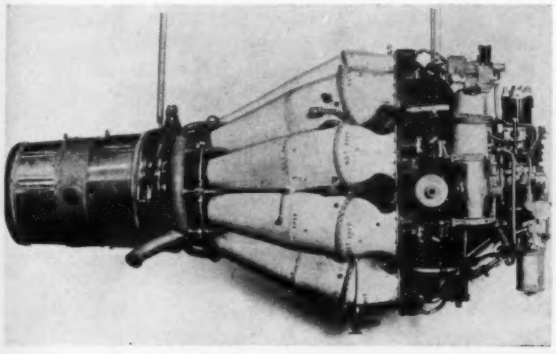

FIG. 41. The Rolls-Royce "Derwent I" engine.

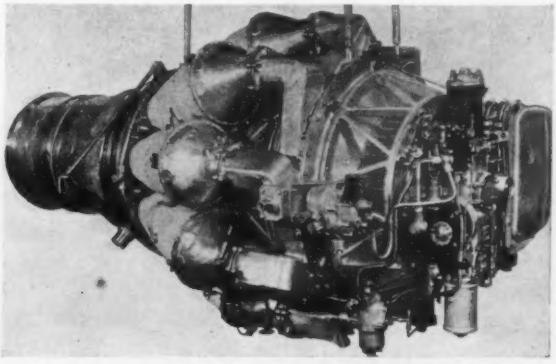

FIG. 44. The Rolls-Royce "Derwent V" engine.

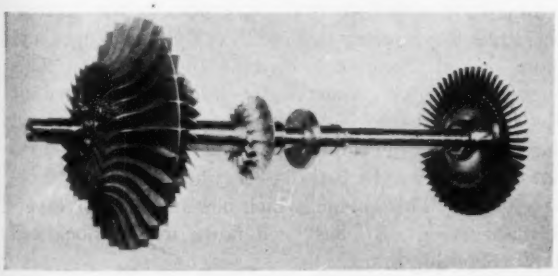

FIG. 42. Rotor of the Derwent I engine.

"Meteor" power-plant production, and the development therefore, on April 1, 1943 (cf. § 3.4). Their efforts in this field, as in the piston-engine field, have been noteworthy. The "Welland" or W.2B.23 engine as fitted in the early Meteor aircraft was rated at a maximum sea-level static thrust of 1,600 lb. (Fig. 40). The "Derwent I" engine (Figs. 41 and 42), now fitted in the Meteor III aircraft, is rated at 2,000 lb. (Fig. 43). The "Derwent V" engine (Fig. 44) embodying the form of diffuser found so successful in the W.2/700, and now coming into service in the Meteor IV aircraft (Fig. 45), is rated at 3,500 lb. (Fig. 46). Naturally, the increases in thrust have not been obtained without increases in engine weight, but the thrust-to-weight ratio has been radically improved from 1.86 for the Welland, to 2.08 for the Derwent I, and to 2.78 for the Derwent V. The Welland (which completed a 500-hour bench test under type test conditions with no replacements apart from a change of flame tubes at 300 hours) and Derwent I engines have

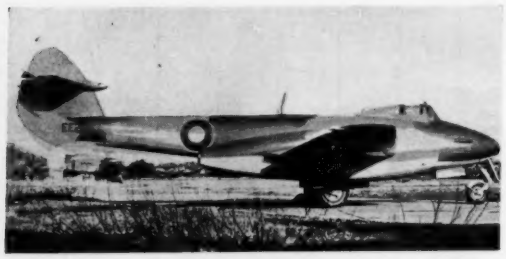

FIG. 45. Gloster Meteor IV aircraft with Derwent V engines.

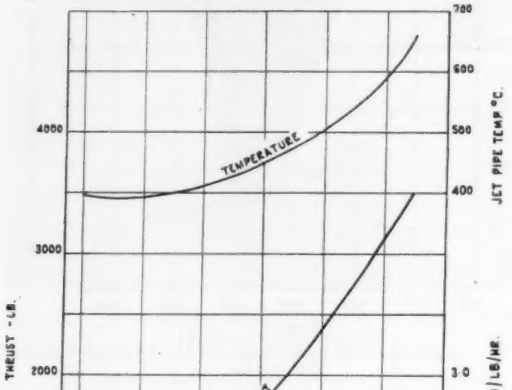

FIG. 46. Sea-level static performance of the Derwent V engine.

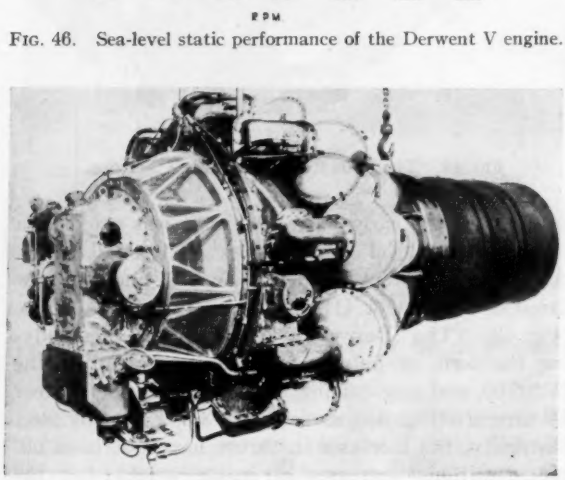

FIG. 47. The Rolls-Royce "Nene" engine.

proved extremely reliable in service and are cleared for 180 hours between complete overhauls; the Derwent V will undoubtedly develop to or beyond that standard.

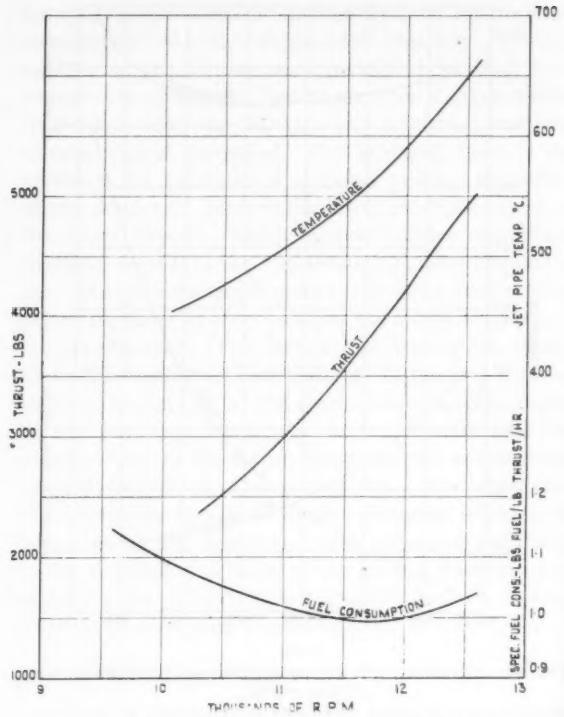

FIG. 48. Sea-level static performance of the Rolls-Royce "Nene" engine.

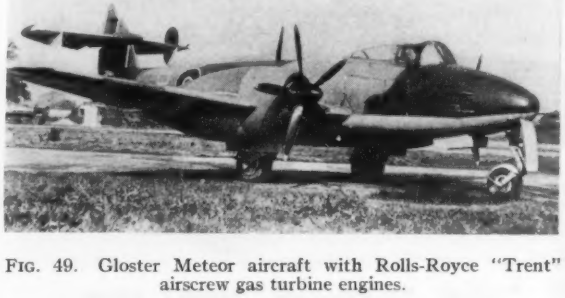

FIG. 49. Gloster Meteor aircraft with Rolls-Royce "Trent" airscrew gas turbine engines.

In parallel with their work on engines for the Meteor, the Rolls-Royce company are developing a bigger and more powerful engine known as the "Nene." This engine (Fig. 47) is geometrically similar to the Derwent V (which, chronologically, is the later prototype) and has been under development for over a year. Its present thrust-to-weight ratio is 3.1, and it still has some way to go. This engine, which has an impressive performance (Fig. 48), has been flown in the Lockheed P-80 (Shooting Star).

Elsewhere in this paper (§ 7.4) I compare qualitatively the jet, ducted-fan, and airscrew versions of the gas turbine. A great many data have to be collected before this comparison can be made realistically quantitative. The first flight experiment designed to this end was entrusted to the Rolls-Royce company, who have converted a few Derwent engines to drive five-bladed

airscrews through reduction gearing. In the new form the engine is called the "Trent" and two Trent engines have been fitted in a Meteor aircraft (Fig. 49). Each has a static performance at sea level of 750 b.hp. and 1,250 lb. thrust. As the Derwent was not designed with conversion to airscrew drive in mind, the Trent is not an ideal airscrew gas turbine. As a means of experiment, however, with all the Welland and Derwent development behind it, it should prove most valuable.

3.7 *The Armstrong-Siddeley Engines*

At the time the Armstrong-Siddeley company were given their first gas turbine contract in November, 1942, there was an overwhelming concentration on the centrifugal compressor type of engine, Metropolitan-Vickers being the only exponents of the axial compressor variety. Armstrong-Siddeley sought to redress the balance and began work on the ASX (Fig. 50), a 14-stage axial flow compressor engine with a two-stage turbine. Unlike the Metropolitan-Vickers F.2 engine, which employs a single annular combustion chamber, the ASX has eleven separate combustion chambers.

Testing on this engine began in April, 1943, and the engine is now giving a maximum thrust of 2,800 lb. in the static condition at sea level (Fig. 51). It is being tested in the Universal Test Bed Lancaster flying test bed (§ 6.3). Tests of the compressor on the Northampton plant (§ 6.2) established an adiabatic efficiency of 87 per cent.

The ASX engine has been adapted to airscrew drive, and in this form, known as ASP, it ran in April, 1945. In the static, sea-level condition it is giving 3,600 s.hp. and 1,100 lb. of jet thrust.

3.8 *The Bristol Engine*

The Bristol Aeroplane company has built an aircraft gas turbine which it has designed especially for long-range medium-speed aircraft. To realize their aim they are using a conventional airscrew driven by an independent turbine and a heat exchanger for heat regeneration

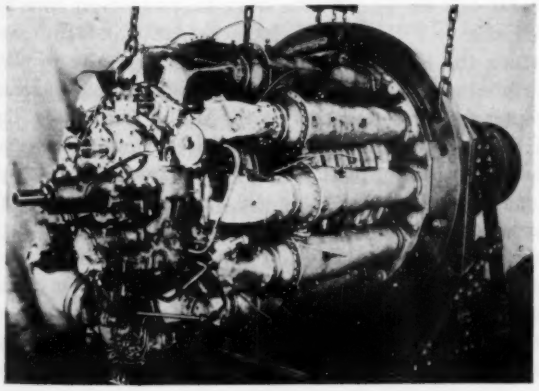

FIG. 50. The Armstrong-Siddeley ASX engine.

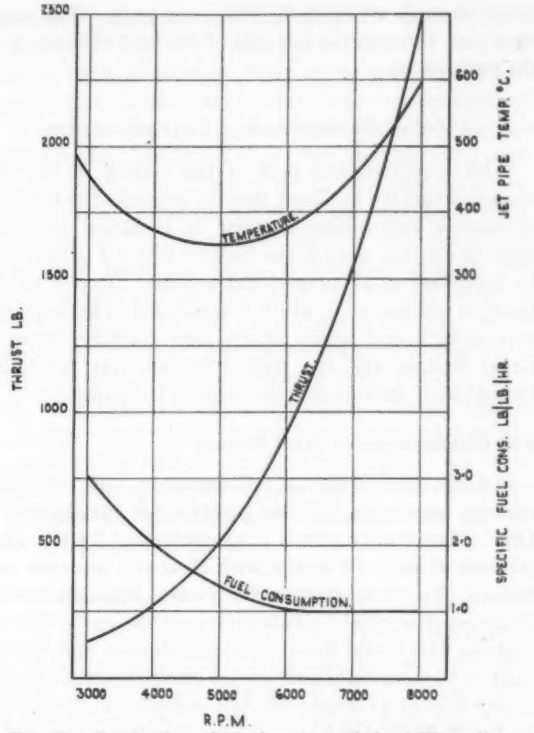

FIG. 51. Sea-level static performance of the ASX engine.

FIG. 52. The Bristol "Theseus" engine.

from the exhaust gases. These features, respectively, provide good propulsive efficiency over a wide range of aircraft speeds and a fuel consumption competitive with that of piston engines of comparable powers. Some 80 per cent of the power of the engine goes to the airscrew, and the rest is used for jet reaction.

The speed of the independent airscrew drive can be varied over a considerable range without affecting overall turbine efficiency.

Briefly, the unit (Fig. 52) consists of an axial-cum-centrifugal compressor delivering air through the heat exchanger to eight combustion chambers and thence to the three turbine stages, the first two of which drive the compressor and auxiliaries and the third the air-

screw through an epicyclic reduction gear. The gases then pass through the hot side of the heat exchanger to the exhaust jet.

4. The Gentle Art of Collaboration

While the technical part of the history of the aircraft gas turbine in Great Britain presents the features of success and failure familiar in technical progress, there is another part of the history which I believe can be described as an unqualified success. I refer to the habit of collaboration which was developed between the several technical teams in my own country, between Great Britain and the United States, and, later, between Great Britain and the British Dominions.

4.1 Collaboration in Great Britain

Strictly, the domestic collaboration falls into two periods, one ending and one beginning in the autumn of 1941. The earlier period is characterized by the associations of pairs of teams with clear-cut common and limited objectives, such as the Power Jets and British Thomson-Houston's collaboration on the early Whittle engines, the Royal Aircraft Establishment and Metropolitan-Vickers collaboration on axial compressors, the Power Jets and Gloster Aircraft company collaboration on the E.28/39 airplane. There is nothing unusual in general character about these bilateral alliances. They are distinguished by pioneering enthusiasms but demand no special comment. In the autumn of 1941, however, there began an experiment in collaboration which has features of interest.

The position in August, 1941, was that Power Jets Limited were actively engaged in the development of the W2B engine; the British Thomson-Houston company and the Rover company were engaged on their versions of the same design; the Metropolitan-Vickers company were working on the F.2 engine; the Royal Aircraft Establishment was collaborating with them and engaged in research and advisory work; Major Halford was in the early stages of the H.1 design; and Ricardo & Company were concerned with governing devices for gas turbine engines; Rolls-Royce were experimenting with the new ideas but had not then settled down to a program.

No special gifts of penetration were needed to realize that, to ensure economy in our efforts to produce power units of the kind as quickly and efficiently as possible, all the parties concerned should collaborate to avoid unnecessary duplication of effort and to pool ideas, testing facilities, and experience.

To achieve this objective, I proposed the formation of a Gas Turbine Collaboration Committee, and a letter of invitation to all the parties I have mentioned to nominate senior representatives to the Committee was despatched on October 3, 1941.

The views of those invited ranged from belief that the Committee might do some useful work to the belief that it would be a wicked waste of time. The same people today are firm supporters of the Committee, which is making a strong bid for vigorous survival in the unfamiliar atmosphere of peace.

The first meeting took place in an M.A.P. conference room in Birmingham on November 1, 1941. Since then the Committee have met roughly five times a year and completed four years' work with its eighteenth meeting on October 12, 1945.

The eighth and thirteenth meetings were held in the M.A.P. in London. All the others were held at the works of the various companies concerned, so that the Committee could see the work in progress and the various gas turbine engines working. It was agreed at the first meeting that each organization would provide, through the Committee, all the others with information on the layouts and performances of the engines they were designing, and there is no doubt that the complete frankness that has characterized the working of the Committee has been a considerable stimulus to individual efforts.

4.2 Collaboration with the U.S.A.

So far as I am concerned, collaboration between my country and the United States in gas turbines and jet propulsion began at 10:30 on July 22, 1941, when Col. (afterwards General) A. J. Lyon of the U.S.A.A.F. and D. R. Shoults called at the M.A.P. for a talk with Air Marshal Linnell and myself. At that time Colonel Lyon was General Arnold's representative in England and Mr. Shoults, normally a senior member of the staff of the General Electric Company of America, was temporarily a representative of the United States Government appointed to help Colonel Lyon in the initial stages of his association with my Ministry on gas turbines.

At that time, Colonel Lyon's duty in relation to gas turbines was to recommend to his superiors what production aid the United States could give to Britain on the Whittle engine and the aircraft using it, and from this beginning, exchange of information on a broad basis has developed.

On that July morning in 1941 I described to Colonel Lyon and Mr. Shoults the work that had been done by Whittle and the Power Jets company and the part which the Rover company were getting ready to play in producing the engines. On July 25, the American representatives, together with Major Brandt of the U.S.A.A.F., Dr. Pye, and I, visited Power Jets and held a discussion with Wing Commander Whittle and his associates. On July 28, the three Americans accompanied me to the Gloster company's works where they saw the two E.28/39 experimental aircraft and met their designer, W. G. Carter. The progress of the F.9/40 twin-engined fighter design, a later version of which became known as the Meteor and was the first allied operational jet-propelled aircraft, was discussed on this occasion.

As a result of these discussions, drawings and data of the Whittle engine were handed to the American officers, and arrangements for a more detailed liaison were made possible by the arrival in England of Major Keirn to assist Colonel Lyon.

This is perhaps the most appropriate place at which to mention the efficient and wholly pleasant liaison which developed between these gentlemen and ourselves. We always remember General Lyon with affection, and his premature death soon after our liaison organization was fully worked out was a sorrow to many of us. I am proud to remember that the first links in the Anglo-American liaison chain on gas turbine work were forged by him and me. Major Keirn followed worthily in his footsteps and he soon became as one of ourselves. His services to the United States have been recognized by his promotion to Colonel and by the award to him for 1944 of the Thurman H. Bane Award. As our association developed, my own deputy, Group Captain Watt, stepped into the role of Colonel Keirn's opposite number, and these two between them have done a classic piece of transatlantic cooperation.

As a result of the August discussions, arrangements were made during September for a Whittle WIX engine (cf. § 2.3) and a set of W.2B manufacturing drawings to be sent to America. This freight, accompanied by Major Keirn and a Power Jets team consisting of D. N. Walker (Chief Test Engineer), G. B. Bozzoni (Experimental Fitter) and Flight Sergeant J. A. King, R.A.F. (attached to Power Jets Limited) left Prestwick on October 1, 1941, in a U.S.A.A.F. Liberator. The aircraft landed at Washington on October 2, and the story of what happened thereafter is American and not English history. I hope it will be related by the protagonists. I will confine myself to remarking upon the more than commendable skill and verve with which the General Electric Company set to work to make the W.2B engine and the remarkable speed with which within one year the Bell Aircraft Company designed and built the twin-engined P-59A engined with it. This airplane, as I have said, though completed after the first Gloster F.9/40, flew several months before it and was the second allied gas turbine type to take the air.

From what I have said so far, it is clear that the initial contacts were with the U.S. Army. On April 2, 1942, however Colonel Lyon brought Commander (afterwards Captain) Kauffman, then newly appointed United States Naval Air Attaché in London, to see me, and from that time the U.S. Navy were in contact with our work. The Commander and Colonel Keirn, then on his second visit to us, attended the sixth meeting of the Gas Turbine Collaboration Committee held on September 26, at the British Thomson-Houston company's works in Rugby. At this meeting the Committee heard from Colonel Keirn a detailed account of American progress in the aircraft gas turbine field.

From these beginnings there came a steady stream of American Service and civilian engineers to England to be let into the secret gas turbine circle, and, as the United States developed their own gas turbine activities, a steady stream flowed in the opposite direction. Whittle visited America in the summer of 1942. American pilots flew British jet-propelled aircraft and British pilots flew American ones. Further details of this liaison would be superfluous here, though mention should be made of the more formal missions which crossed the Atlantic. First there was a U.S. Navy and Army Air Forces Mission during July, 1943; next a U.S. Navy Bureau of Aeronautics Mission during September, 1943; and then a British Special Projects Mission to the U.S.A. during May, 1944. All three missions were successful technically and socially, and in England we have pleasant recollections of associations which were all too brief.

4.3 Collaboration with the Dominions

As is generally known, the tremendous expansion of British engineering activities under the war stimulus resulted in a shortage of technical talent. This shortage was acutely felt in the aircraft gas turbine sphere. As a result, an arrangement was made between the British Government and the governments of Canada, Australia, South Africa, and New Zealand by which service and civilian technical personnel came to England to help the work forward, obtaining at the same time training in the new art. This arrangement, with its reciprocal advantages, has worked admirably. The first men arrived in England in March, 1944, and only now is the scheme coming to an end. The men first of all received an intensive course of lectures and training on hack engines at the Power Jets works and then passed out to the various collaborating firms, many graduating ultimately to positions of considerable responsibility.

A special relationship has developed with Canada, where, in July, 1944, Turbo-Research Ltd. was formed. This company, owned by the Canadian Government, is a research and development organization on the model of Power Jets (Research and Development) Limited and the two companies maintain a useful liaison. In the winters of 1943–1944 and 1944–1945, British engines were tested by the Canadian company under cold conditions on special test beds near Winnipeg, where the winter climate is particularly suitable for the purpose.

5. Aspects of Development and Research

The development histories of the engines I have mentioned exhibit many common features, and I propose to deal briefly and generally with some of the problems encountered.

5.1 *Bearings*

Bearings were the source of a number of failures and of discussions on the Collaboration Committee. Ball, roller, plain journal, and Michell bearings all had their adherents. It would be dangerous to suggest that a general solution was found, but success is associated with seating designed to withstand mounting distortions, with the correct (and considerable) diametral and axial slackness in ball bearings and limitation of bearing temperatures by careful cooling. The majority of designers incline to ball bearings, some preferring roller bearings at the turbine end of the shaft. Lubrication has proved less difficult than was expected in the early days, and "solid" lubrication, as distinct from lubrication with an atomized spray, with oil of relatively low viscosity, is the common practice.

5.2 *Sheet Metal*

All the engines so far have employed sheet-metal construction for the bulk of the combustion system, and this sheet metal was at one time a prolific source of failures. These resulted from inadequate experience of resistance welding, from fatigue and fretting due to high frequency aerodynamic buffetting, and from oxidation through contact with flame or distortion due to bad temperature distribution.

The first type of failure was eliminated by study of and experience in resistance welding, the correct choice of material, and careful attention to finish.

Fatigue and fretting failures have been eliminated by removal or diminution of the sources of vibration, by stiffening or other forms of redistribution of material, by the careful design of corners and junctions, and by locating components to allow freedom for expansion while retaining sufficiently accurate positioning. In certain designs, the problem of fretting has been dealt with by arranging small air gaps at the susceptible junctions; this has had no ill-effect on performance.

Failures from flame or poor heat distribution were common in the flame tubes of the early Whittle-type engines, but they have almost disappeared now that combustion chambers have been developed with no thin metal parts in the flame region and the control of the fuel injection has been improved. The annular combustion chamber, however, is still subject to thermal distortions and cannot yet be regarded as satisfactory.

5.3 *Impellers*

A serious crop of impeller failures was experienced as the engine ratings were increased. Investigation showed that in the early engines these had been avoided more by good fortune than by good judgment, as it was discovered that the failures were attributable to resonant vibrations excited by aerodynamic impulses from the diffusers. After much theoretical and empirical work, designs were evolved which proved to be satisfactory (cf. § 3.1). From these, more precise data have been obtained and impellers can now be successfully designed with confidence.

The achievement of a satisfactory impeller was delayed through pilots and test personnel instinctively avoiding the resonant frequencies because they found the corollary noise to be uncomfortable. Moreover, the test schedules approved by the Ministry of Aircraft Production for most designs did not, by sheer chance, demand any prolonged running at speeds which were afterwards found to be critical. This explains the curious fact that an impeller to a design now known to be unsatisfactory has been known to last 200 hours, whereas another to the same design has failed in less than 5 hours.

In the history of the elimination of mechanical failures in gas turbine engines, the solution of the impeller problem is an outstanding achievement.

5.4 *Turbines*

The development of the appropriate high-temperature steel for turbine blading at the Firth Laboratories under Dr. Hatfield was one of the major factors which made the aircraft gas turbine possible, and the development of the engine has been accompanied by developments in the materials of construction. At one time, turbine blade failures were unfortunately common. Today, they are a rarity. Accuracy in manufacture, avoidance of small radii at root junctions, sound stressing methods, understanding of turbine vibration problems, as well as improvements in material, have all played their parts in the elimination of failures.

The outstanding blading material developed for the British engines is Nimonic 80. This material was discovered by the Mond Nickel Company in 1940, soon after they had discovered Nimonic 75, which is particularly suitable for combustion parts. Nimonic 80, with its excellent creep and fatigue properties at the working temperature, is sufficiently easily forged at about 1,100°C. to permit blade blanks to be stamped practically to finished size. It is now the standard material for turbine blading in British engines.

A number of failures of turbine discs and rims has been experienced, but now the likelihood of failures of this kind is small. This is due not only to improvements in the design of gas seals and the arrangements for disc cooling, and in the provision of better materials, but to the extension of knowledge of stress effects. In the gas turbine disc, the presence of high temperature and steep temperature gradients means that high stress may be accompanied by plastic strain. The stress distribution departs from the predictions of elastic theory, there is high residual stress when cold, and the natural vibration frequencies are functions of the stress conditions. Also, the cycles of strain and temperature modify the stress-strain characteristics of the material, in which there may be structural changes. The Gas Turbine Collaboration Committee determined upon a

thorough exploration of the disc problems and appointed a special subcommittee* for the purpose. In addition to theoretical investigations, valuable experimental research is being done by Dr. Bailey in the research department of the Metropolitan-Vickers company to explore quantitatively the phenomena I have noted.

For this work a special electrically heated furnace (Fig. 53) is used to produce required stress distributions by applying known temperature gradients. This demands air cooling at the inner parts of the disc. To maintain conditions symmetrically in the disc it is slowly rotated. Thermocouples are inserted in the disc at different radii and their leads brought out to slip rings. Vibration-measuring equipment can be applied to the apparatus to measure natural frequencies under temperature gradients.

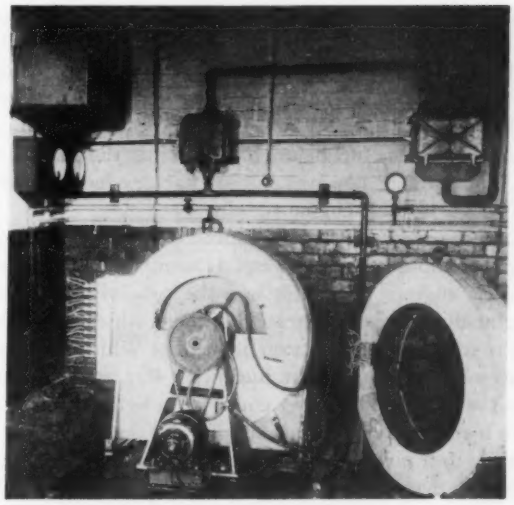

FIG. 53. Metropolitan-Vickers electrically heated furnace for turbine disc research.

5.5 *Integration of Research and Development*

This is an example of a piece of research springing directly from development and illustrates the absurdity of trying to define too precisely the boundaries between research and development. It is also illustrative of the close linking we have always been able to maintain between the development engineers and the research teams. Insofar as it shows research depending from development however, it is typical only of a class of cases. Properly, research has anticipated difficulty as often as it has resolved difficulty.

Each gas turbine firm has had its own group of workers in research or, probably more accurately, in that broad belt wherein research and development are indistinguishable. In addition, there has been at all times a large group engaged in general problems, correlative and advisory work. This is the group, led in recent years by Hayne Constant, whose pioneer work in axial compressors at the Royal Aircraft Establishment I have already noted. Today they form part of the organization known as Power Jets (Research and Development) Limited.

The broad treatment of a large canvas, which at the outset I indicated as my objective, while it should do rough justice to engineering achievements, is less suited to an appreciation of the work of the research teams. I may perhaps mention, however, without being tempted into the details which they invite, the researches by Mond Nickel, Firth Vickers, Jessops and the National Physical Laboratory on high-temperature materials; the work by the Power Jets team on burners; the work by the Power Jets (Research and Development), Lucas, and de Havilland teams on combustion technique; the work of the Asiatic Petroleum company and the Combustion Panel on the physics of combustion; and the work by Constant's team on gas analysis, on blade vibration, and on blading design.

From this last group, I propose to mention as an example of general experimental research of the kind designed to anticipate difficulty, the work done in collaboration with Armstrong-Siddeley on the effect of axial spacing of blading. Whereas when the early axial compressors were being designed (§ 2.2) we thought it necessary to allow, between fixed and moving blade rows, an axial distance equal to at least one-third of the blade chord, now, as the result of experiment with a compressor in which blading spacing can be varied, we conclude that the closer together the blade rows are spaced the better. It is ironical that the old compressor "Anne" was originally designed with close spacing, which was abandoned upon knowledge obtained of contemporary Swiss practice (cf. § 2.2).

5.6 *Combustion*†

The story of combustion development deserves some special notice, as, of the many problems presented by the design of the aircraft gas turbine, the combustion problem was probably initially the most critical. Although there was as a background an established technique of oil burning, the special problems of the aircraft gas turbine demanded a new approach, since the engine's performance is extremely sensitive to the pressure loss through the combustion chamber, and above all the weight and volume of the chamber must be small. There was no question of starting development with a spacious brick-lined chamber, such as is used for raising steam. From the outset, combustion systems have been schemed in light-gage sheet metal and, in addition to efficiency and pressure loss considerations, prevention of overheating and mechanical failure were matters of prime consequence.

* Dr. S. L. Smith, Dr. R. W. Bailey, Dr. Ker Wilson, Dr. J. F. Shannon.

† British and American combustion terminology is different. I believe that the British "burner" corresponds to the American "nozzle" and the British "flame tube" to the American "burner."

The original Whittle unit was built with one large combustion chamber, and the early trials were done with liquid injection, the burners being of the low dispersion pencil type. Combustion was unsatisfactory, and at an early stage Whittle introduced the technique of vaporizing the fuel before injection into the chamber; for some time, however, unsatisfactory combustion continued to limit engine running.

Looking back, two events stand out in the story of the development of successful combustion for the Whittle engine. The first was the subdivision of the combustion into ten separate chambers, which appeared in the second reconstruction of the first Whittle engine. The great advantage obtained with this arrangement was that a single chamber could be individually tested with a relatively small air supply and developed on a test rig to a point where it could be applied to the engine with every chance of success. The second notable event was the advent of the burner, due to I. Lubbock of the Asiatic Petroleum company, which gave a finely atomized spray of liquid fuel over the wide range of output required. Although with the vaporizers which were embodied in the unit as reconstructed with ten separate combustion chambers the combustion itself reached a relatively satisfactory stage, development was held back by the mechanical failure of the vaporizing apparatus, chiefly because of overheating and blockage of the vaporizing tubes which were contained in the flame tube and subjected to irregular heat distribution. The introduction and development of the Lubbock burner quickly led to the first set of combustion equipment to allow engine running of any duration. The engine for the first flight of the E.28/39 aircraft was equipped with Lubbock-type burners operating in a swirl-type combustion chamber developed by Power Jets Limited, and this system formed the basis of most of the subsequent combustion system designs.

During these early days, the Royal Aircraft Establishment had been doing work on the combustion problem and developed a type of chamber based on the original low dispersion jet, injecting into the air in an upstream direction. This was later taken over by Metropolitan-Vickers for their annular chamber, and a combustion system of this type is now being used in the F.2 units.

When the Rover company was engaged in the production of the W.2 type engine, Messrs. Lucas, who were supplying the combustion equipment, started a combustion test department which has since grown and now does a considerable part of the combustion development and supply of components for several manufacturers of aircraft gas turbines. Lucas's chief contributions have been in the development of the so-called "straight-through" multichamber combustion system.

With the interest in combustion problems spread to several centers, the Gas Turbine Collaboration Committee set up a subcommittee, known as the Combustion Panel, in July, 1942. This was under the Chairmanship of Professor Lander of the Imperial College of Science & Technology and was composed of combustion specialists from Asiatic Petroleum, Lucas, Power Jets and the Royal Aircraft Establishment, together with members from the universities and research organizations. This Panel provided means for consultation, for laying down general lines of research on common problems, and for guiding this work and preventing overlapping. A large portion of their work has fallen into a study of the properties of injectors in respect of design data and measurement of performance and, in particular, the measurement of the particle sizes of the atomized spray. Other investigations coordinated are a study of carbon formation and methods of mixing. The work of the Panel has been valuable in supplying a common meeting place for those engaged in combustion, and the size has been increased to include several advisory members from associated fields.

The state of combustion is, now, generally satisfactory, the main requirements of pressure loss, efficiency, absence of carbon formation, and stability of operation having been achieved. An increase of life is a desirable improvement, and future advances will be toward further reductions of pressure loss and greater heat release rates to allow reductions in size and weight of the chamber. The majority of British engine designs are still based on individual chambers with downstream atomized injection and with air admission utilizing various degrees of swirl. Two exceptions are the Metropolitan-Vickers engine, which has an annular chamber with upstream pencil injection, and the Armstrong-Siddeley unit, which is based on vaporization and partial mixing with air before burning commences. No completely satisfactory annular chamber has yet been designed, and it appears probable that, although annular air casings may appear, the inner flame tube will still be in the form of a number of discrete units.

6. Facilities

As has been made clear, gas turbine research and experiment started in a modest way, and the experimental facilities available in the early days were few and elementary. For their early work on cascades of airfoils Harris and Fairthorne used that now common device the cascade wind tunnel, but for a long time after their work with it no special apparatus for gas turbine investigation came into existence. The testing of the early small axial compressors required little that was special, and it was found possible to test the larger compressors of the B.10, D.11, and F.2 schemes by means of a steam turbine which the Metropolitan-Vickers company had built for a power station. The attack on the combustion problem, however, initiated the demand for special facilities.

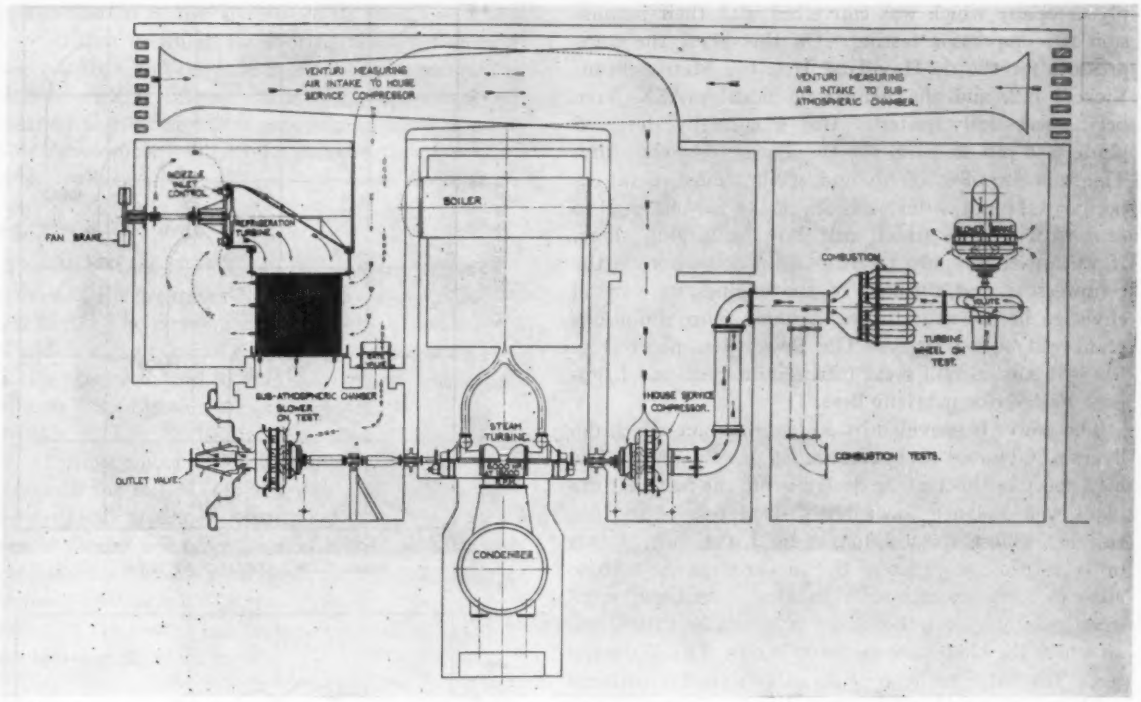

FIG. 54. Diagrammatic layout of Power Jets' 6,000-hp. plant for component testing.

6.1 Combustion Test Plant

In the cases of engines in which the combustion system consists of a number of separate chambers, a great deal can be done by experiment upon a single chamber with air at approximately atmospheric pressure. For such work, the Power Jets company use 45-hp. test plants, each able to deliver $3^1/_2$ lb. of air per sec. at 3 lb./in.^{2} gage pressure; these plants are fairly typical and are similar to those used in the early combustion tests for the Whittle engines. When, however, it becomes necessary to test a system consisting of a single annular chamber, as in the case of the F.2 engine, or to test even a single chamber from a multi-chamber system at higher pressures, much greater powers are required. Some of the earliest work at engine pressures on single chambers from the ten-chamber system of the Whittle-type engines was done with compressors which had been built as part of the facilities for creating a new tunnel under the Thames near Dartford. In the Power Jets company, such work has been done for some years past with the air supply from two Browett & Lindley compressors of 600 hp. each, which, as operated at present somewhat below their maximum output, provide 3 lb. of air per sec. at 40 lb./in.^{2} gage pressure. Such sources of air supply are, however, inadequate for the work ahead, and it is hoped to create far more generous sources of air supply to meet the demands of aerodynamic, as well as of combustion, testing.

6.2 Component Test Plant

It was realized in the early days of jet propulsion engine design that, whereas the overall efficiency of a gas turbine engine could be determined by measurements of the characteristics of the engine as a whole, reliable deductions of the efficiencies of the three main components—the compressor, the turbine, and the combustion system—from such engine tests were not possible. Furthermore, there were clearly serious difficulties in improving the individual efficiencies of the separate components when they could only be tested as part of a complete gas turbine engine. The need for means of testing the main components separately was evident. Equally evident was the necessity for providing quite appreciable power for the purpose, for even in the small W.2/700 engine, which gives a thrust of some 2,200 lb., the power which the turbine has to supply to drive the compressor is about 4,500 hp. Even for engines of this relatively small size, therefore, a power station giving something in excess of 4,500 hp. was necessary.

We have made several shots at the problem; I have mentioned the temporary use of a steam turbine by Metropolitan-Vickers. In addition, Rolls-Royce built a compressor testing plant driven by one of their Vulture engines. In these cases the power was insufficient, however, and it was necessary to throttle the intake air. A more powerful plant was a power station of about 6,000 hp. belonging to the Northampton Electric Sup-

ply company which was converted with their permission to compressor testing. On this plant the compressors for the de Havilland H.1, the Metropolitan-Vickers F.2, and the Armstrong-Siddeley ASX have been successfully tested. Also a specially designed plant was put in hand for the Power Jets company. This was also of 6,000 hp. and, while a most useful experimental plant, will clearly shortly be far too small to deal with engines which will soon be coming along. If we are thoroughly to investigate and improve the compressors and turbines of the engines we can all envisage in the near future, a much more ambitious plant will be necessary. The Power Jets plant (Fig. 54) is in any case of great intrinsic interest, and I propose to describe it briefly here.

The power is provided by a steam turbine which delivers 6,000 hp. at its maximum r.p.m. of 18,000. The high speed of this turbine derives from the fact that the plant was designed to deal with compressors and gas turbines whose natural maximum r.p.m. are of this order. In consequence of the high r.p.m. the turbine appears to be exceptionally small. In fact the maximum diameter over the blades is 13 in., and the drum on which the blades are mounted is only 8 in. in diameter. The rotor really consists of two similar turbines made to opposite hands with their exhausts facing and joining into a common outlet branch.

Steam is supplied to the turbine by a Lamont marine-type boiler.

A drive can be taken from the turbine at either end. With one drive shaft in position the turbine drives the compressor under test in an airtight chamber. With this drive shaft removed and the one at the other end of the steam turbine inserted, the turbine drives a slave compressor the duty of which is to supply air for the testing of combustion systems or turbine wheels.

In the case of compressor testing, altitude conditions can be simulated. The way in which this is done is of particular interest. Air is led through a venturi measuring air intake over the tube surfaces of a heat exchanger and, thence, passes into an air turbine. The work done by the air in passing through the air turbine is dissipated by means of a fan which the air turbine drives. Having done this work in the turbine, the air is cooled and the cold air passes through the tubes of the heat exchanger into the cell containing the compressor under test. It is because the compressor is consuming air that the air is drawn through the apparatus in the way I have described.

It should be noted that the incoming air, passing between the tubes cooled by the air discharge from the air turbine, deposits its moisture in the form of snow on the outsides of the heat exchanger tubes. The air delivered to the compressor under test is consequently appropriately dry, though the humidity in the test cell does not necessarily correspond with the standard atmosphere pressure which has been produced therein. The control of the pressure is achieved by varying the nozzle

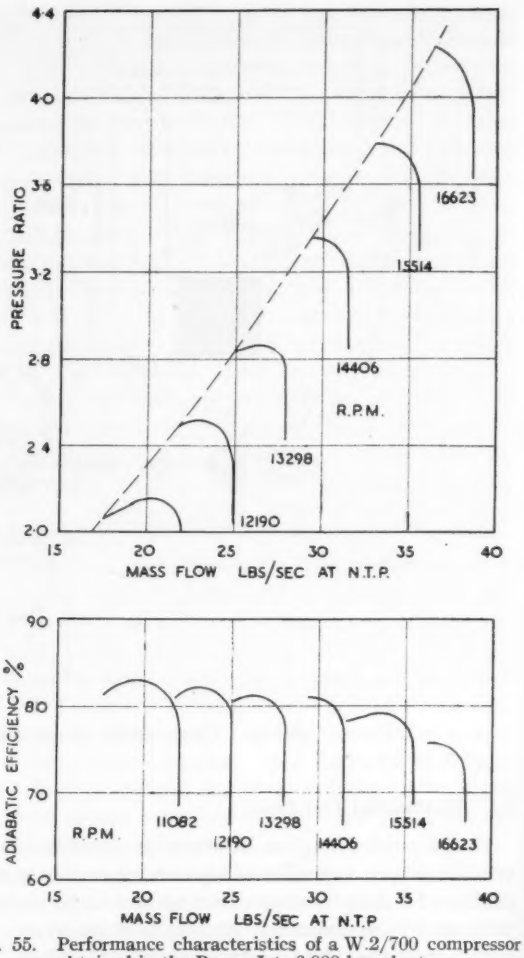

FIG. 55. Performance characteristics of a W.2/700 compressor obtained in the Power Jets 6,000-hp. plant.

area of the air turbine, altitude conditions being simulated by small nozzle areas and, consequently, reduced mass flows to the pressure cell. This refrigerating plant is extremely convenient and is capable of cooling the incoming air by 34°C.

The plant, then, is capable of testing a compressor (Fig. 55), a turbine, or a combustion system or supplying air for special tests. A similar function is performed by a rather smaller plant which Power Jets (Research and Development) Limited have installed at their Pyestock station. This is essentially a 4,000-hp. compressor driven by electric motors drawing their power from the grid system. Primarily, it is intended to supply large quantities of air for aerodynamic and thermodynamic tests. This air can be used, appropriately heated, to drive turbines, and so a variety of experiments can be made with the plant as the basis. The three plants so far mentioned are in the charge of the Government-owned Power Jets (Research and Development) company and are at the service of industry as well as being used for general investigations. The Metropolitan-Vickers company has been able to adapt a

power station in the vicinity of its works in a similar way to that in which the Northampton power station was adapted, and they are therefore well equipped to do development testing on their own compressors. It is clear to us that any firm seriously engaged in gas turbine development will need a plant of this general character.

6.3 Flying Test Beds

Another item of experimental apparatus which has proved of the greatest value to us is the so-called flying test bed. The convenience of having a large airplane in which a gas turbine engine can be mounted, its behavior observed, and its fuel system and ancilliary apparatus tested under varying conditions of speed and altitude, is clear. The first of our flying test beds was a Wellington bomber which was modified to take a W2B engine in the tail. This first flew with its gas turbine engine under test in August, 1942. This aircraft and a second similar one have been used by the Rolls-Royce company for development work on their Whittle-type engines. A third Wellington flying test bed (Fig. 56) is used by the Power Jets company for flight experiment on engines of the same type.

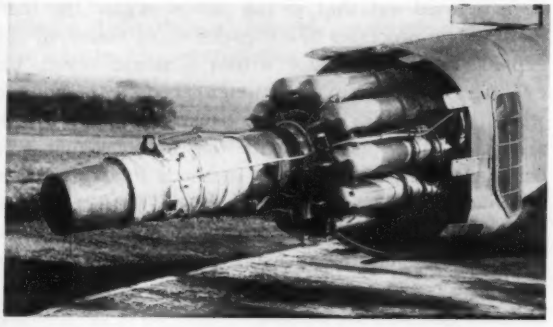

FIG. 56. Vickers-Wellington flying test bed for centrifugal compressor gas turbine engines; the Power Jets W.2/700 engine is installed.

The second of our flying test beds was a Lancaster bomber* from which the tail turret was removed, thereby permitting the mounting of a Metropolitan-Vickers F.2 at the rear end of the fuselage. A large air intake was mounted on the top of the fuselage and fortunately proved not to interfere with the efficiency of the tail unit. The first flight with the F.2 under test was made on June 29, 1943, and a great deal of testing on this engine and others of its class has been done in the Lancaster bomber first used and the later Lancaster bomber* which replaced it (Fig. 57).

Another converted Lancaster bomber, somewhat grandly called the Universal Test Bed Lancaster, has

* The first Lancaster used was the original prototype Lancaster and was replaced by a production model as soon as possible because of its nonstandard characteristics and the consequent difficulties of maintenance.

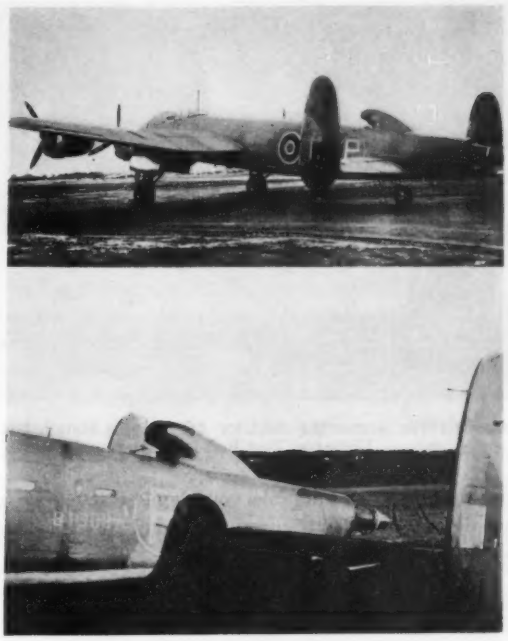

FIG. 57. Avro Lancaster flying test bed for axial compressor gas turbine engines; the Metropolitan-Vickers F.2 engine is installed.

recently been delivered to Power Jets (Research and Development) Limited (Fig. 58). In this aircraft the engine under test, instead of being mounted in the tail, is mounted in what was previously the bomb bay. In this position a variety of engines of different forms can conveniently be slung. The first to be installed was the Armstrong-Siddeley ASX (Fig. 59).

The information gathered from the flying test beds has been of the greatest value, and it is intended to continue with this method of experiment. Although only relatively low speeds are possible, the degree of instrumentation it permits and the convenience of having on board staff with no duties other than testing the gas

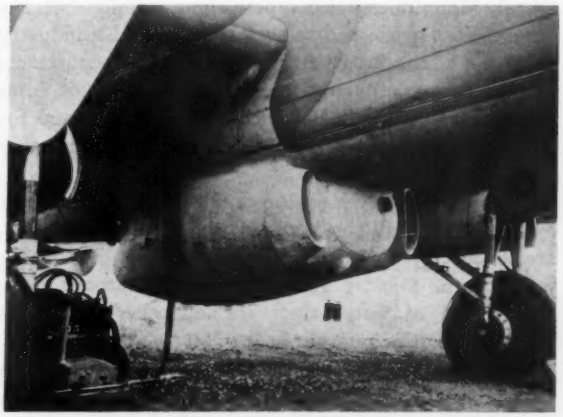

FIG. 58. Universal Test Bed Lancaster; the Armstrong-Siddeley ASX engine is installed.

FIG. 59. The Armstrong-Siddeley ASX engine installed in the Universal Test Bed Lancaster.

turbine make it a valuable complement to experiment and development with aircraft dependent for sustentation on the engine under test.

7. THE CONTEMPORARY SCENE

7.1 General Organization

If the traditional bird took one of his more comprehensive views now of British gas turbine work, he would see a well coordinated activity. In the midst of it is the Ministry of Aircraft Production. Therein, overall policy is determined and implemented by contract. Grouped around the Ministry are the Government-owned Power Jets (Research and Development) Limited, conducting investigations in the general interest, and the Rolls-Royce, de Havilland, Bristol, Armstrong-Siddeley, Metropolitan-Vickers and British Thomson-Houston companies developing engines along lines indicated by the discussion I have given of their past achievements.

All these companies are doing their gas turbine work under Government contract. The Power Jets company however, with its 100 per cent Government shareholding, does not compete commercially with the others; it is concerned to serve them, British industry at large, and the British fighting services. It constructs engines for experimental purposes but does not enter into production. Production for the Services and for civil air transport purposes is the business of the other companies.

Associated with the engine companies, and with their own contacts with the Ministry of Aircraft Production, are the aircraft firms. I have mentioned a little of the work of the Gloster and de Havilland aircraft companies, and, of course, there are now others whose gas turbine activities have not been made public.

The forum for intimate technical discussion and collaboration is the Gas Turbine Collaboration Committee, though it is now possible to discuss the broader gas turbine problems in open sessions of The Royal Aeronautical Society and the Institution of Mechanical Engineers.

As the comprehensive vision of our observant bird is unlikely to be highly selective, he would perceive, in addition to these activities in the aircraft world, the awakening of heavy industry to the possibilities of the gas turbine. Extra-to-aircraft work, however, is outside the scope of this paper, and its only relevance lies in the fact that much of the experience gained in developing the aircraft gas turbine is valuable to gas turbine workers in other branches of engineering, and this experience is being made readily available to them.

The air-frame and engine firms have generally appreciated the revolutionary character of the significance of the gas turbine in their industry, and technical men are working out the consequences of its introduction.

7.2 Union of Air Frame and Engine

One interesting viewpoint is that the design of gas turbine aircraft drives home a lesson imperfectly appreciated and, indeed, less important, in the days of the piston engine—the lesson that there are not two units, air frame and engine, but one unit, the aircraft, in which air frame and engine are interdependent and to be blended without bias. Elaborating this point, it has been pointed out that in the piston engine the thermodynamic processes of compression, combustion, and expansion all take place within a single vessel, the cylinder; that the maximum weight economy is achieved when the several cylinders are packed closely together; and that the piston engine is, in consequence, a compact mass, installational methods for which have become more or less standardized. In the gas turbine, on the other hand, the three main thermodynamic processes take place in three different components—the compressor, the combustion system and the turbine—and not only are the shapes of these three components susceptible of considerable variation but their disposition in relation to one another can also be varied. The number of final possible forms is indeed legion, and with all this variation at his disposal it is pointed out that the aircraft designer has a wonderful opportunity of compromising between and interrelating the structural and power-plant parts of his machine. Considerable lip service has been paid to this point of view, but up to the present I am unaware of any aircraft in any country having been designed in a way consistent with the point of view I have reviewed. Logically, it would require that the whole design of the aircraft, air frame and engine came under one man, and no arrangements for this kind of design control exist in Great Britain. However, we all think this is a good idea.

7.3 Forms of Compressors

In this matter of the integration of air-frame and engine design no dissenting voice has been lifted. There are, however, other aspects of the gas turbine

airplane on which opinion is less unanimous. As might be expected from the way in which our aircraft gas turbine work had its origin, there tend to be axial and centrifugal schools of thought. There was a time when the whole matter was seen very simply by mediators who propounded that the centrifugal compressor engine was the lighter but that the axial compressor engine by virtue of its higher compressor efficiency was the more economical in fuel. Ergo, one used the centrifugal engine for short-range airplanes and the axial compressor engine for long-range airplanes. The matter is, however, not so simple. Not only do the axial compressor experts foresee more economical methods of construction but the centrifugal compressor experts foresee higher compressor efficiencies. For my part, I refuse to take sides and am merely grateful that there should be possible compressors of such fundamentally different form, making still greater the number of geometrical variations at the disposal of the aircraft designer. In making comparisons of the various kinds of engines, however, it is essential to remember that bench performance figures do not necessarily provide sound criteria. Installational differences and differences in the internal aerodynamics mean that the variations of performance with speed and altitude are in general functions of the type of engine. As yet, there are not sufficient flight data to provide correlation between flight and bench performances. In any comparison of single- and double-sided compressor engines, on which I have already commented from a geometrical standpoint (§ 3.5), this must be borne in mind.

7.4 Propulsive Systems

The respective roles of the jet propulsion gas turbine, the airscrew gas turbine, and the ducted fan gas turbine cannot yet be said to have been delineated. It may be that clear lines of demarcation between their spheres of influence are not possible; so much depends upon the duty to be performed by the aircraft in which a gas turbine is to be installed. It may be worth while to elaborate this trite observation.

For the highest sustained subsonic speeds, the jet propulsion engine is supreme. But only rarely will speed alone be the criterion. Generally, economy will be a more important consideration. Consequently, a great deal of attention is paid to fuel consumption, usually expressed in terms of lb./lb. thrust/hour or lb./b.hp./hour. On such criteria, the jet propulsion engine generally appears inferior to the airscrew and ducted fan engines and has sometimes been associated, in consequence, with short endurance aircraft. So simple a comparison is, however, rarely relevant. If pay load, be it men, mails, or missiles, is to be a maximum, then, roughly, power-plant weight plus fuel weight must be a minimum. If, for the moment, we accept power plant plus fuel weight as the criterion, we may assume for purposes of illustration that, in a given set of circumstances, a value of the criterion may be defined such that for greater values the airscrew turbine gives greater ranges than the jet turbine, and for smaller values the opposite obtains. At the critical value, the airscrew turbine will be the heavier and more expensive plant, and the airplane will be slower; on the other hand, the volume of fuel carried will be smaller and the height of operation modest. The jet propulsion engine will be cheap in first cost, and the airplane will be fast; the fuel volume will be large and the airplane will have to operate in the region of the tropopause. The choice will depend upon the relative importance in the particular case of first cost, fuel cost, speed, and operational height, and no general conclusion seems likely.

There are two glosses on this argument worth mentioning. One is that the gas turbine is relatively insensitive to fuel quality and low grades can be used. The other is that, although I have implied that the jet propulsion gas turbine is most economical at altitude, it *may* be that, even for ranges so short that climb to great height is not feasible, the lightness and simplicity of the engine and the cheapness of the fuel will make the jet propulsion installation the most economical choice.

I am aware of the simplifications I have indulged in, but I do not think they affect the validity of my illustration of the theme that the correct form of engine to choose depends upon the job to be done.

When all this has been said, however, it is still true, I think, that for the longest ranges required the jet propulsion gas turbine is beaten by the airscrew gas turbine. In the long-range sphere the competition may be between the airscrew and the ducted fan engines. Once again, plain fuel consumption would be a misleading criterion. Weight of power plant plus fuel for the journey is still the more appropriate. Its significance in this comparison is less than in the previous one, however, because the disparity in power-plant weight is less and another factor, which I have in my rough argument heretofore ignored, may prove decisive. This is the aircraft drag. Comparing the tractor airscrew gas turbine aircraft with the ducted fan gas turbine aircraft, the slipstream drag of the former may outweigh the better fuel consumption of its engines. If pusher airscrews are advanced to clinch the argument in favor of the airscrew version, then discussion becomes complex, as the mass balance of the aircraft, installation problems (e.g., jets fouling airscrews), and engine design (e.g., airscrew shaft and exhaust at the same end of the engine) are rapidly introduced.

Whatever the result of this competition, it would be wrong to assume that either airscrew or ducted fan gas turbines were essentially long-range engines. I have already indicated that the airscrew and jet propulsion variants may compete for short ranges, and the high thrust and small frontal area of the ducted fan engine

may well make it particularly appropriate to fast climbing fighters. Even though a jet propulsion fighter might be faster than a competing ducted fan fighter, the latter would probably climb the faster.

7.5 *Influences of the Gas Turine on Aircraft Design*

It would seem reasonable to expect that the great variety of power-plant form offered by the gas turbine, the improved performances which it makes possible, and the possibilities of indulging the reciprocal influences of air frame and power plant will result in notable changes in the design of aircraft. I feel that these changes must be of four kinds—

Changes in the current convention forced upon the aircraft by the requirements of the gas turbine.

Changes in the current convention representing improvements permitted by the gas turbine.

Changes dictated by new operating conditions made possible by the gas turbine.

Unorthodox layouts harmonious with the gas turbine.

In the first class are the large holes required in the front of the aircraft for the entry of air, the large holes in the rear for the emission of the exhaust, and the elevation of the tail plane in certain known aircraft designs to avoid the exhaust jets.

Another design change which the gas turbine enforces in airplanes designed to operate from restricted spheres, such as carrier decks, arises from the nature of the thrust variation of a jet propulsion engine. An ordinary landing approach is made with the engine throttled back. Should the landing be baulked, the throttle is opened up, but, as some 5 sec. is required to attain full throttle from idling r.p.m., the operation is liable to be hazardous in a baulked deck landing without some special device. The special device with which experiments are being made by Power Jets is a thrust spoiler (Fig. 60). This is, in effect, a pair of swing doors which, when fully open, leaves the jet unrestricted but which, when closed, deflects it in such a way that the net thrust reaction is negative. Intermediate positions of the doors may, of course, be used if required, to vary the thrust at constant r.p.m. The use of the device is obvious. Instead of reducing r.p.m. in the landing approach, the spoiler doors are partly or wholly closed. If the landing is baulked, the doors are swung open in a split second and the full thrust is immediately available. So far, the best braking thrust obtained is 12 per cent of the thrust with the jet unimpeded.

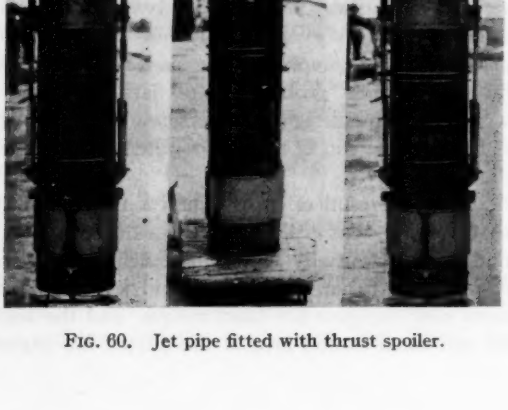

FIG. 60. Jet pipe fitted with thrust spoiler.

In the second class there is, for example, the shortening of the undercarriage permitted by the jet propulsion gas turbine and the improved opportunities of attaining the ideal of absorbing the power plant entirely into the wing.

Changes of these two kinds are exemplified in current aircraft, and they have not produced designs of markedly different outline from those to which we have become used.

Under the third heading the major changes are likely to be consequences of the extremely high speeds which the gas turbine, particularly the jet propulsion gas turbine, makes possible. The need to defer compressibility as long as possible introduces violent sweep forward or sweep backward of leading edges, and wing sections will become more exaggerated. There will be greater insistence than ever on smooth surfaces, with a consequent reaction on structural method. All these changes, however, are consequences of the demand for high speed, however produced, and are consequently to be less directly associated with the gas turbine than changes under the other three headings.

Turning to my fourth heading, it may be that the general scheme of the airplane as we know it—long fuselage, wings in front, tail behind—is the best for all possible power plants. Some bold spirits have taken leave to doubt this, however, and have put the tail in front and the wings behind, or left the tail off altogether and shortened the fuselage. Some still unrepentant eccentrics[4] have advocated the so-called "flying wing." While there is no well-defined tendency to depart from orthodoxy along any of these lines, there is a suspicion that the gas turbine should be carefully considered in relation to them, and some case for arguing that, however unprofitable their opponents may have proved them to be with piston engine power plants, the case is altered with the gas turbine.

For example, if in the large airplane case it is admitted that the engines should, if possible, be absorbed without trace in the wing; and if the gas turbine, with its possible small frontal area, can be so absorbed; then as, in so modifying the orthodox layout, the center of gravity must move far aft, will there not be a most serious balancing problem? Further, would not the need to arrange for the exhausts to miss the fuselage or the tail elements be avoided if neither fuselage nor tail was present? Further still, if as so often happens, desire for smooth airflow over the wings forces the designer to pusher airscrews and if, as happens as fre-

quently, problems of ground clearance then become a positive hindrance, may not some radical change be necessary? I do not know the answer to these questions, but I know they are being studied and I have a sneaking hope that unorthodoxy may triumph.

There is another aspect of design that is being debated. Is there a case for the large airplane? Its chief merit was not that it carried a great many people at once, but that it was capable of providing amenities and even luxuries during a long journey. But, we ask, will there be any long journeys? In flying from London to New York, shall we need even any food? It would appear rather that the problem will be how to avoid the embarrassment of having two luncheons, one before leaving and the other on arrival. High speed and high frequency seem to work together against the large airplane, and the strongest case appears to be with those who claim that nothing more than railway comfort will be required on any journey.

7.6 Comparison with German Effort

When data on the German gas turbine engines began to be available, there was, of course, keen interest in comparing the enemy's technical achievements with the standards reached in Britain and the U.S.A. At an early stage in our investigations of the German engines it was clear that, despite the fact that the flight trials of the Heinkel-Hirth jet propulsion gas turbine aeroplane in August, 1939, had antedated the flight of the E.28/39 by 19 months, our designers had outstripped their German rivals. True, the German production of operational gas turbine aircraft was well ahead of the Allied effort, but the Luftwaffe had taken delivery of engines which the British and American air forces would have found completely unacceptable. Their short running lives and inadequate reliability were a serious embarrassment in service. To compare the latest Allied engines with the German operational types would be unfair, but comparison of contemporary British and German operational engines is sufficient to demonstrate that on the basis of thrust/weight ratio and fuel consumption the German engines were outclassed.

Engine Type	Thrust (lb.)	Thrust / Weight	Fuel Consumption (lb./lb. Thrust/Hour)
Welland	1,600	1.86	1.12
Jumo 004	2,000	1.20	1.40
Derwent I	2,000	2.08	1.18
BMW 003	1,760	1.31	1.40

This clear-cut triumph does not blind us to the facts of ingenuity in design and manufacture exhibited in the German engines, nor to the merits of the experimental and testing facilities with which the enemy had equipped himself. We are studying his methods and his equipment with interest and advantage. We are impressed with the Munich plant for testing jet propulsion gas turbines under high-altitude conditions and are indebted to the United States for affording us facilities to test some of our engines in it.

We are also impressed with the scale and equipment of the laboratories devoted to gas turbine research. The ratio of the useful work done with this equipment to the scale of it, is, however, ditsinctly unimpressive, and I attribute this to the poor liaison between the engine designers and the research workers. In some fields of endeavor the German organization of technical resources was undoubtedly good, but in the gas turbine field I conclude that it was indifferent.

8. Conclusion

Such is my picture of the British contribution to gas turbine progress. As I gave warning at the outset, it is incomplete and I am conscious of its inadequacy. I hope, however, that it will give some idea of the nature and scope of British work toward an engineering revolution, the fruits of which, I pray, may promote the peace, prosperity, and brotherhood of us all.

9. Acknowledgments

I wish to record my grateful thanks to a number of individuals, in the organizations mentioned in the paper, who have helped me in gathering the data for it.

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Discussion of the Lecture

Dr. Hugh L. Dryden, Chief, Mechanics and Sound Division, National Bureau of Standards, presided as chairman for the presentation of the Lecture. In a brief introduction Dr. Dryden explained that the Lecture is endowed by The Vernon Lynch Fund, established in 1937 by the late Edward C. Lynch in memory of his brother. It is delivered one year by a guest from abroad and the next year by an American. Dr. Dryden then expressed the good fortune of the Institute in having Dr. H. Roxbee Cox as the 1945 Lecturer.

At the conclusion of the Lecture the following prepared comments on the subject were given.

Col. D. J. Keirn, Power Plant Laboratory, Engineering Division, Air Technical Service Command: "I am happy to be present to hear Dr. Roxbee Cox give us such a splendid picture of British aircraft gas turbine development, and I feel greatly honored to be permitted to offer some comments.

"I have been involved in the development of gas turbines in America since the Summer of 1941. My role has been largely one of liaison in which I have had the opportunity of observing much of the work in the United Kingdom, as well as follow quite closely the work carried on in this country.

"It is hardly appropriate for me to attempt to fill in any of the American picture of gas turbine development, but I feel that I should comment on the collaboration that existed in England, between our country and the United Kingdom, and collaboration as it existed between manufacturers and Governmental agencies in the United States.

"My observations of the degree of collaboration between the firms in England lends emphasis to Dr. Cox's statement of the unqualified success of such collaboration. The great progress made in the few years of war and the present excellence of several British gas turbines could not have been achieved but for the wholehearted way in which various firms interchanged "know how" through the medium of the collaboration committee.

"Correlation of efforts in the United States was in general achieved by liaison between the development agencies of the Army and Navy and the manufacturers involved.

"Although a degree of collaboration was achieved through the medium of Dr. Durand's subcommittee of the N.A.C.A., created to study the problems of jet propulsion for aircraft, collaboration between all interested firms was not achieved in this country for several reasons.

"The major consideration was perhaps the necessity of limiting the scope of gas turbine activities because of the production burden carried by the majority of aircraft-engine manufacturers. It appeared undesirable to bring them into the picture and divert their engineering efforts from the work of getting their own equipment whipped into shape for production and war service.

"Another consideration was the high order of secrecy under which the gas turbine work was conducted and which rendered more difficult the dissemination of information that might otherwise have stimulated research by more people. Needless to say, much time has been lost by not informing and creating the interest of more people, especially manufacturers of accessory equipment.

"Another factor has been the reluctance of some manufacturers to share their knowledge with others for fear of compromising their future position.

"Collaboration with the British has, on the other hand, been most beneficial. Although certain gas turbine projects were initiated in this country prior to the establishment of liaison with the United Kingdom, such liaison has made possible a greatly accelerated development program due to encouragement by their successes, the exchange of visits by technical people, and the exchange of technical information which, irrespective of its value to the United Kingdom, has been of inestimable value to us.

"Dr. Cox's comments in regard to combustion are of particular interest. The vast amount of excellent work done in the United Kingdom is recognized, as well as the general excellence of combustion in their turbines on test. We in this country are still experiencing combustion trouble, not so much on the test stands as in flight under abnormal conditions—for example, idling and starting conditions at high altitude and apparent poor combustion efficiency at high altitude. We wonder if Dr. Cox's optimism really infers that a solution is in sight for satisfactory combustion under all probable conditions of flight.

"Associated with the gas turbine combustion problem is combustion in the ram jet, which is apparently a more difficult problem and one even further from solution in this country.

"I cannot refrain from making some comment on the remarks of Dr. Cox in regard to the union of air frame and engine. Many in this country also share the point of view that the gas turbine and the air frame should make an integrated whole, the airplane. However desirable this may appear, many practical considerations became evident whenever an attempt is made to weld more closely the turbine to the air frame.

"Two major factors mitigating against the achievement of the ideal appear at the present time. First is the time element involved in bringing an airplane from initial design to flight-test stage as compared with the time required to bring the gas turbine, not to initial test, but to flight-test stage. Perhaps past experience, wherein aircraft designed around certain engines under development were necessarily compromised by installation of different engines when the intended engine ran into development snags, has caused the procuring agencies to become cautious.

"Secondly, the mechanical design and manufacturing techniques employed by the turbine manufacturer and the air-frame manufacturer are widely different, despite the fact that certain fundamental sciences are becoming more common to both. Although this consideration does not preclude close coordination of airframe and turbine design, it requires a mutual understanding of the problems peculiar to each and must, of necessity, involve compromise on the part of both airframe and turbine designer. Complete integration of

the turbine and air frame must be approached with caution. Should the integration be carried so far as to permit of common structure, as when structure of the turbine becomes as well a stressed portion of the structure of the air frame, many quite weighty objections may be raised.

"I should like to conclude my comments by saying that I feel that Dr. Cox's summary of British gas turbine development is most excellent and timely. I know of no other man who has contributed so much toward the coordination of efforts in gas turbine development and who has done more to establish the delightful relations that have existed between the British and American workers in the gas turbine field. I hope that these relations may continue and that we may be of mutual help in the development of this new means of aircraft propulsion."

Carlton Kemper, Executive Engineer, Aircraft Engine Research Laboratory, National Advisory Committee for Aeronautics: "The members of this audience have grown up with the reciprocating engine. You have seen the reciprocating aircraft engine power plant grow from the 12 hp. of the Wright Brothers' first engine to approximately 4,000 hp. in 1944. Dr. Cox's statement that research facilities capable of testing compressors and turbines of 6,000 hp. are already obsolescent will come as a shock to many of you when you realize that the jet engine is still in its infancy. As Dr. Roxbee Cox has stated, the jet engine is the greatest revolution since the Wright Brothers made their historical flight 42 years ago.

"It is fortunate indeed that the Institute of the Aeronautical Sciences has been able to persuade Dr. Roxbee Cox to give us the historical background of the jet engine in Great Britain. One is amazed when he realizes that all the necessary research and development work to bring the jet engine to its present stage of development in Great Britain has been carried out during the time the war has been fought. This is an outstanding example of how research conducted even during war times may 'pay off' in either war or peace. Evidently the British did not make the same mistake as the Germans who drafted into the Army large numbers of scientific workers. It was only when the result of scientific work conducted by the Allied physicists, chemists, and engineers was brought to bear on the Germans in the form of radar, high-octane-number fuels, night fighters, and improved pursuit and bomber airplanes that they realized their mistake and worked frantically to get their scientists out of the Army and assigned to the universities and research laboratories.

"Dr. Roxbee Cox pointed out the important role played by Colonel Keirn as one of the leaders of the Army group that brought the British jet engine to America. Dr. Cox stressed the extreme importance of having research facilities available for testing individually the compressor, combustion chamber, and turbine for the jet engine. He pointed out that Whittle started out to design, construct, and test the individual components of the engine but had to give up this idea because of the lack of research facilities. He constructed and tested the jet engine as a unit. To secure best operating efficiency, the individual components should be tested separately to ensure that the requirements of the turbine and compressor are matched.

"The National Advisory Committee for Aeronautics has constructed at its Aircraft Engine Research Laboratory at Cleveland a 20-ft.-diameter altitude wind tunnel capable of testing aircraft power plants complete with the nacelle, cowling, superchargers, radiators, heat exchangers, and the flight propeller, if any. This tunnel is designed to operate at an air speed of 500 m.p.h., at altitudes from sea level to 50,000 ft., and at air temperatures to −47° F. The tunnel has a makeup air capacity of 100 lbs. of air per sec. The fuselage and the two jet engines used in the Bell P-59 airplane were the first jet engines tested in the tunnel for the Army Air Forces. Since that time almost every jet engine designed for the military services has been operated under flight conditions in this tunnel. One of the important results of the research in the Altitude Wind Tunnel was the discovery that the performance characteristics of combustion chambers at flight altitudes could not be successfully predicted from performance tests of combustion chambers at sea-level conditions.

"Dr. Cox pointed out the short period of time required to increase the power output of the jet engine. To double the power output of a modern reciprocating aircraft engine and to develop it to the state in which it could be specified as a power plant for service airplanes would require the expenditure of approximately $5,000,000 and require $2\frac{1}{2}$ years of design and testing. The Rolls-Royce company has been able to design, construct, and start the testing of a new jet engine in 152 days. The jet engine can be readily scaled up in size without encountering any major design difficulties.

"The curves in the paper show minimum specific fuel consumption at sea level for all of the jet engines, except the Rolls-Royce 'Nene,' to be approximately 1.1 lbs. fuel per lb. thrust per hour. With the 'Nene' engine the specific fuel consumption is 0.98 lb. fuel per lb. thrust per hour. The lower specific fuel consumption of this engine is probably due to the reduction in pressure loss with the straight-through combustion chamber and to an improvement in the combustion efficiency. Further reduction in the specific fuel consumption of jet engines will result from the fundamental studies of combustion now in progress in this country at the universities and research laboratories.

"Improving the performance of the jet engine is primarily a problem of increasing the operating efficiency of the compressor, the combustion chamber, and the turbine. Improving the metallurgical properties

of materials for combustion-chamber liners, turbine discs, and blading will increase not only the efficiency of the unit, since it will permit operating at higher temperatures and higher compression ratios, but also the life of the jet engine.

"No one has stated a limitation as to the maximum thrust output of the jet engine. Heretofore, the aircraft designer has been able to use the most powerful reciprocating engine that the manufacturer has produced. For cruising conditions with the jet engine the maximum range may be obtained by cutting out individual engines. This may serve as a limitation to the size of the jet engine but, even with this limit, the size will probably be not less than 10,000 lbs. of thrust. This *is* a revolution in aircraft power-plant design."

R. P. Kroon, Manager of Engineering, Aviation Gas Turbine Division, Westinghouse Electric Corporation: "Dr. Roxbee Cox's paper is of particular interest to us to-day because, as far as I know, this is the first time that we in the United States get a comprehensive picture of the British gas turbine developments.

"Most of the American designers, like myself, did not get acquainted with our British jet propulsion friends until about 2 years ago. We have seen glimpses, but it was left to Dr. Cox to give us the complete survey, all the more interesting because most of the participants are still walking this earth.

"Dr. Cox presents a fascinating saga of engineering development in a new field, with the trials, tribulations, and triumphs that seem to be a part of all pioneering.

"One realizes the vast scale on which this pioneering took place. It involved the engine industry, the aircraft industry, the Ministry of Aircraft Production, government laboratories, and private concerns for component and material development. No longer can a lone inventor, or even a small team, be expected to solve the many new problems that come up in developing a radically new article for service operation.

"Instead, we see in Britain and in the United States, as well as in Germany, a new pattern of industrial development in which the efforts of many contributors are combined.

"In this connection the Gas Turbine Collaboration Committee (which Dr. Cox states will continue to function after the war) is of interest. It goes to prove that within industry it is possible to have a great deal of exchange of information and coordination of activities without giving up individuality of design features. It appears that most designers are sufficiently convinced of the superiority of their own brain child to resist accepting what seems to be the common denominator.

"A tendency toward standardization *is* apparent in the selection of materials for those parts that are subjected to high temperature. While the American scene has been decorated with a great many forged and cast alloys with intriguing names, the British restricted themselves to the use of relatively few forged alloys. The materials selected were good and did the job.

"All in all, one concludes that the Gas Turbine Collaboration Committee served the British well in guiding the development and in avoiding duplication of efforts.

"Dr. Cox mentions new forms that the gas turbine might take. The idea of 'compromising between and interrelating the structural and power-plant parts of the (aircraft),' as he calls it, has, also in this country, appeared as a dream that so far has not become real. We have had visions of gas turbines built in the shape of wing sections and of jet engines of which the combustion chambers, mounted on the side, would be a more or less integral part of the airplane. However, such arrangements lack flexibility and it appears today that the aircraft industry is better served with standard engines which can be built in several configurations. We feel that, for example, the Westinghouse design, in which accessories can be mounted either at the top, the bottom, or the side of the engine, is a rational step in the right direction.

"There is no question but that the next few years will show further advances that will make the jet engine a more flexible power plant and which will integrate the aircraft and the engine more completely. There will be thrust destroyers or "spoilers" as the British call them. There will be thrust augmentation, probably in several forms. The jet engine will be taken advantage of for cabin supercharging, cabin heating, and deicing. Its exhaust will be used to give directional control to planes going at extreme speeds.

"In discussing the relative values of the jet engine, the ducted fan, and the gas turbine propeller drive, we should remember that the jet engine, as built so far, has been a 'limit design' to suit the high-speed fighter plane. The capacity of the engine—the thrust—for a given size and weight was deliberately made as high as possible. The jet velocities are high. This means that for any but ultra-high flight speeds the propulsive efficiency of the jet is poor.

"But, if one chooses to operate with lower exhaust velocities (and, therefore, better propulsive efficiencies) one can obtain an attractive combination of performance and weight. Such a 'derated' engine, operating at low turbine inlet temperature, would have long life. It would combine the inherent simplicity of the jet engine with the efficiency and weight characteristics of the ducted fan. I, personally, feel that this type of jet engine has a real future in the transport field.

"While our acquaintance with the British gas turbine developers is only of recent date, the relation has been a friendly and a beneficial one. Both sides appreciate that they can learn much from each other. It is my sincere hope that these good relations may continue."

Donald F. Warner, Assistant Designing Engineer, Aircraft Gas Turbine Engineering Division, General Electric Company: "Dr. Roxbee Cox has presented a comprehensive and excellently detailed statement of the history and present status of the Aircraft Gas Turbine industry in England. The generous disclosures on performance and the numerous illustrations of apparatus and test facilities make it an attractive paper to the engine and aircraft manufacturers and one that will be studied thoroughly in the industry and in the services.

"Like the other discussors, I have been favored with a preprint of the paper. This I have studied carefully with a view to discussion here. The paper is written in a vein that does not encourage controversy, and it is most pleasing and instructive to read.

"Sections 7.4 and 7.5 in the paper summarize the hopes and expectations of the Aircraft Gas Turbine enthusiast and therefore constitute a most important content in the paper. There is one consideration, however, that is lost in the comparisons in that no mention was made of the possibilities of the conventional engine mechanically compounded with the turbosupercharger; in other words—the compound engine. At this time, it is fitting to state that I, too, am a gas turbine enthusiast, but, in the interest of seeing the whole story presented, I cannot avoid calling attention to the possibilities of the compound engine as a future power plant also. Were it not for this perhaps discordant note, today would be a field day for the gas turbine folk. It may be, however, that the conventional reciprocating engine is not quite so dead as the dodo and that resource to compounding the conventional engine will give that type of power plant a lease of life for unknown years to come. It may also be that the economies made possible by compounding will establish a target that will not be attained (or even approached) by the gas turbine except through the medium of complex and weight-involving cycles. This coming form of power plant places new emphasis on the need for higher and higher efficiency on the part of the competing gas turbine-propeller combination, and this may be of the order that would confine its future consideration to only those installations in which powers involved are beyond the capacity of the then existing conventional compounded engines to produce.

"There is a further influence to increase the spread in speeds available between propeller and jet aircraft. In the current work on airfoils of the swept-back or swept-forward type, many developments will come about which should improve the propeller and the wings of aircraft. But the gains in speed made possible by utilizing this means would tend in the end to allow greater speed increases in jet applications than would be expected in conventionally propelled craft. This is so since, in propeller craft, the limitation on speed will again be established by performance of the propeller tips which, if they are enabled by improvements to approach or exceed with good efficiency the velocity of sound, would be used in craft with wings operating at much less than that speed; the jet-propeller aircraft, however, should take full advantage of the gains to be made possible by this type of airfoil. The situation in the future will remain as now—that where speed is the prime requisite, the jet's the job.

"I should like to take this opportunity of expressing a compliment to our British cousins on their great accomplishment in the field of gas turbines. To have engaged in such widespread activity in all the branches of gas turbine work, in centrifugal and axial, jet, and propeller applications simultaneously, and to have paralleled these activities in many agencies creates a load that perhaps only those who have been exposed to this industry can imagine. There are a myriad of problems in any one of these endeavors to be encountered and solved. The results obtained speak highly for the foresight of Dr. Cox in establishing his Collaboration Committee under which a spirit of mutual cooperation was instilled into the British enterprise."

Effective Control of Stalling Characteristics of Highly Tapered and Swept-Back Wings

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SUMMARY

A tested new method of airfoil selection is conceived to assist the designer in overcoming present hazardous stalling tendencies on highly tapered (3:1 or more) and swept-back (15° or more) wings, and to control stall at inception and through progression. This practicable method eliminates high drag penalties and other undesirable characteristics that develop with washout and highly cambered wing tips when employing two controlled sections.

Three controlled sections, one located at the wing root, another at a mid-span station, and the third at the wing tip, are connected by straight lines. The principal parameters affecting the maximum section lift coefficient, viz., the section thickness ratio and camber, are chosen to satisfy the section lift coefficients required by the computed span-load distribution at the Reynolds Numbers of the three spanwise control stations.

The resulting spanwise distribution of maximum lift coefficients permits the designer to exercise close control over the progression of the stall from its inception and thus reduce washout and camber variation to a minimum. A small but appreciable increase in maximum wing lift coefficients is also obtained.

SYMBOLS

b	=	wing span
C_l	=	section lift coefficient
C_L	=	wing lift coefficient
C_{l_i}	=	ideal or design lift coefficient of an airfoil section (see definition in N.A.C.A. T.R. No. 383)
t/c	=	thickness/chord = relative airfoil section thickness ratio
$y/(b/2)$	=	spanwise station (fraction of semispan)
ϵ	=	spanwise washout of zero-lift lines

REASONS FOR THE STUDY

THE NEED TO OVERCOME hazardous stalling characteristics of highly tapered and swept-back wings selected for other than purely aerodynamic reasons has given rise to the present study.

An investigation of the fundamental reasons for these unsatisfactory stalling tendencies reveals that high plan-form taper and sweep-back of the wing create three unfavorable effects on the stalling characteristics:

(1) A highly tapered plan-form and/or a large degree of sweep-back lead to a deviation from the elliptical span-load distribution in the direction of higher lift coefficients at the wing tips for a given wing lift coefficient (Fig. 1).

(2) The decrease of chord length from the root to the tip reduces the Reynolds Number and, hence, the maximum lift coefficient attainable for the sections in the outboard wing panel.

(3) The outwardly directed spanwise cross-flow observed on many highly swept-back wings is an additional incentive for airflow separation near the wing tip.

These three unfavorable developments have been universally counteracted by two measures:[1]

(1) Aerodynamic washout, that is, washout of the section zero-lift lines, produced by twisting the tip chord with respect to the root chord.

(2) The employment of a more highly cambered airfoil at the wing tip than at the wing root.

For manufacturing simplicity the corresponding airfoil stations of the root and tip sections are customarily connected by straight lines. The resulting spanwise variation of aerodynamic washout, camber, and thickness ratio is hyperbolic inasmuch as they vary as

$$y = (a + bx)/(c + dx)$$

where, a, b, c, and d are constants depending upon wing geometry, x is the spanwise station, and y is the variable to be determined (aerodynamic washout, camber, and thickness ratio, respectively). Typical spanwise variations are shown in Fig. 2.

The principal effect of washout consists of a reduction in the section lift coefficients at the wing tip and an increase of section lift coefficients inboard, as shown in Fig. 3. The resultant improvement in the stalling characteristics, however, is gained at a small penalty in induced drag through the prevalence of positive and negative basic lift over the wing span at zero wing lift.[2] Washout does not change the section maximum lift coefficients attainable at the various spanwise stations.

Camber and thickness variations do not affect the span-load distribution (if their slight influence on the section lift-curve slopes is disregarded), but they do modify the spanwise distribution of the maximum attainable section lift coefficients.

The straight-line variation of airfoil chord results in a linear decrease of the Reynolds Number from wing

Received December 21, 1944. To have been presented at the I.A.S. Thirteenth Annual Meeting, which was canceled to cooperate with a Government appeal (January, 1945) to refrain from holding conventions and meetings. Revised and resubmitted July 11, 1945.

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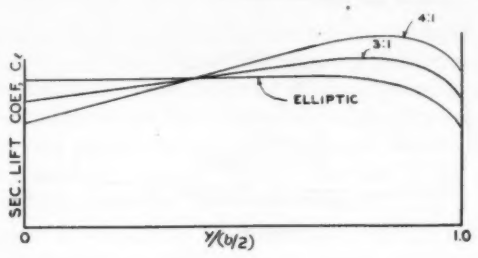


FIG. 1

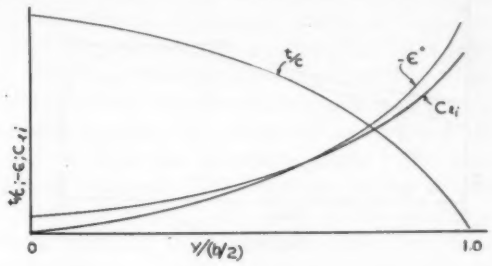


FIG. 2

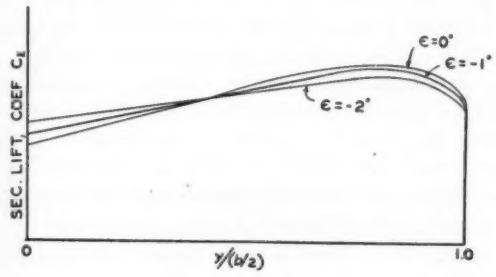


FIG. 3

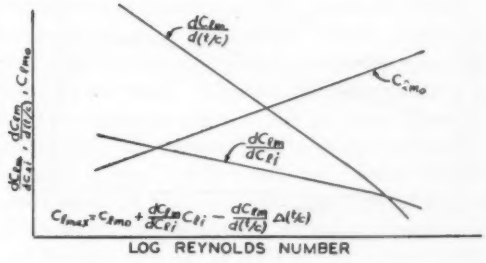


FIG. 4

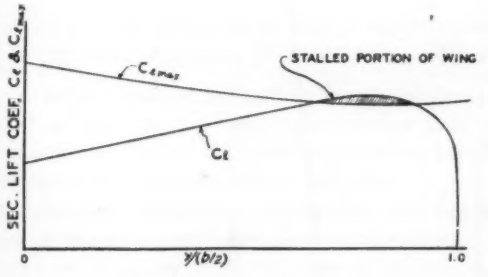


FIG. 5

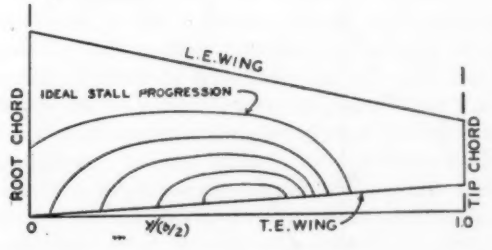


FIG. 6

root to tip. A reduction of section maximum lift coefficients along the span, for a given airfoil section, ensues consequently from the typical maximum-lift variation with Reynolds Number shown in Fig. 4.

A typical spanwise variation in section maximum lift coefficient resulting from the linear fairing of a wing root section and a more highly cambered wing-tip section is portrayed in Fig. 5. It is evident that the line of maximum lift coefficients is concave upward and may even have intermediate stations below the two extremes, because the favorable effect of camber (and thickness) following a hyperbolic law is insufficient to compensate for the unfavorable effect of the linearly diminishing Reynolds Number.

As a rule, the resulting stall pattern is unsatisfactory for any but the lowest taper ratios and may become critical for taper ratios in excess of 3:1 and for highly swept-back wings (see Fig. 5). The stall inception close to the wing tip and the comparatively slow progression of the stall farther inboard produce the most undesirable type of stall, with little or no warning, violent rolling moments, and neutral or unstable pitching moments through the stall.

Any attempt to improve these unsatisfactory stalling characteristics of highly tapered and swept-back wings by flattening the actual span-load distribution through aerodynamic washout, or by raising the curve of the available maximum lift coefficients near the wing tips through adequate amounts of mean-line camber, or by both of these measures, introduces a large drag penalty. In addition, the span-load distribution at the high lift coefficients occurring during pull-outs and steep turns

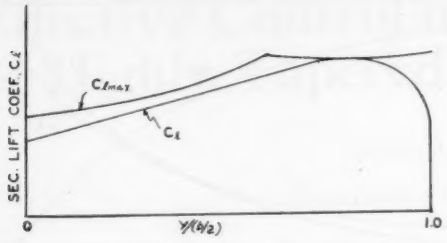


FIG. 7

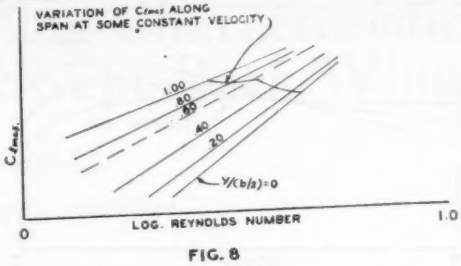


FIG. 8

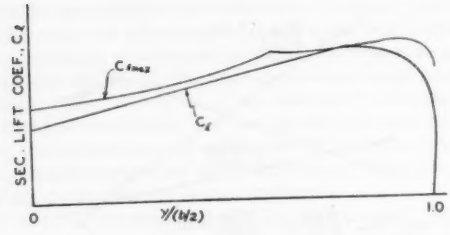


FIG. 9

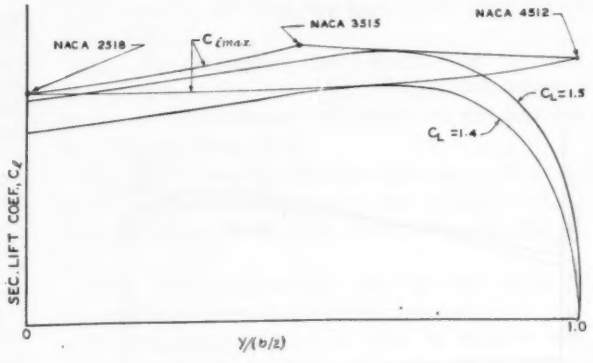


FIG. 10

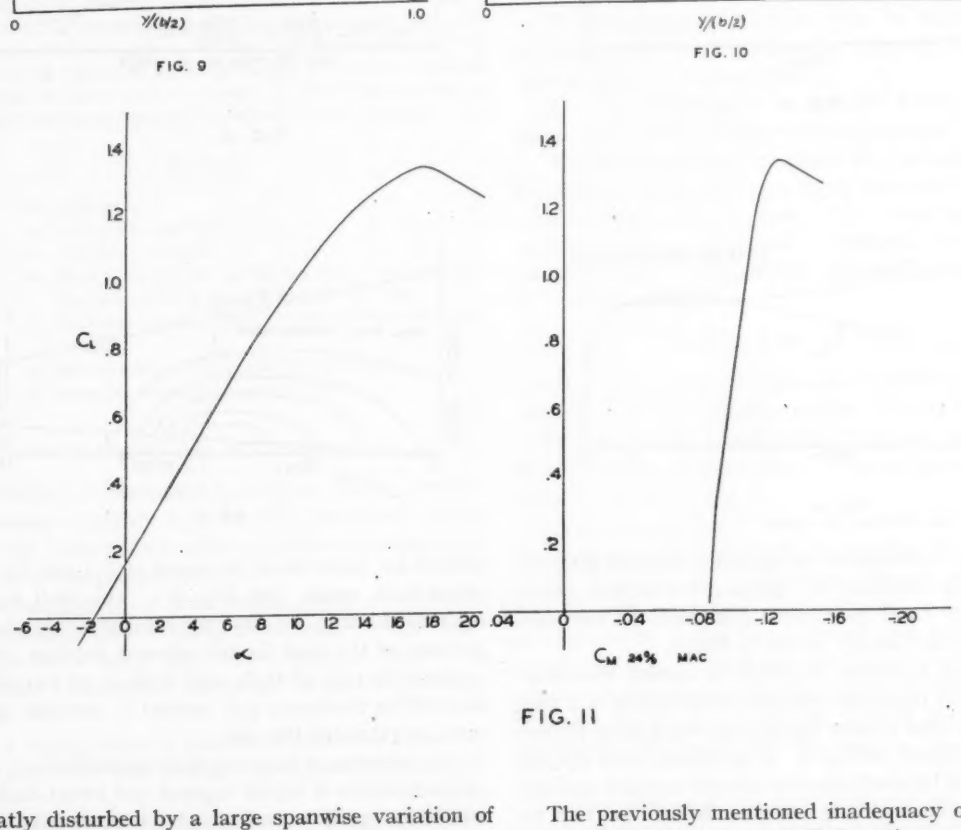


FIG. 11

is greatly disturbed by a large spanwise variation of camber. The peak pressure coefficients at high section lift coefficients increase more rapidly over the sections with small camber than over those with large camber and result in a premature shock stall at the inboard sections, followed by an outboard shift of the air load and a consequent increase in the wing bending moment.

The previously mentioned inadequacy of the linear taper with only two controlled sections on highly tapered and swept-back wings has led to the development of wings with three controlled sections to permit the designer to obtain the desired stall inception and progression with a minimum of washout and camber variation.

Definition of Desirable Stalling Characteristics

From the pilot's point of view a desirable stall is preceded by a gentle but reliable warning in the form of a mild tail shake some 5–10 m.p.h. above stalling speed. The stall should be free from sudden roll, aileron snatch, or severe premature tail buffeting, and should be accompanied by a rapid negative increase of the static longitudinal stability derivative, dC_m/dC_L.

In order to achieve these desirable characteristics it is advocated that stall separation should start approximately at mid-span, outboard of the horizontal tail (to prevent premature tail shake), and should spread, fairly evenly, inboard and outboard, without involving, however, any portion of the wing tips prior to the full breakdown of the flow at the wing root (Fig. 6). The tail shake then coincides with the decrease in the lift-curve slope at the approach to the lift-curve peak. The rapid yet gradual spanwise spread of the separated area, simultaneously, prevents the formation of a deep local stall in a chordwise or vertical sense at any section. Steep, spanwise pressure gradients and, hence, spanwise cross flow are thereby effectively prevented.

The inboard expansion of the stalled area, aside from producing the desired stall warning, will reduce the downwash at the tail; the increased static longitudinal stability and lowered trim C_L provide the nose-down pitching moment that is required for prompt recovery after the stall.

Stall Characteristics of Wings with Three Controlled Sections ("Tri-Section Wing")

The subject method is based on the use of three controlled sections, at the wing root, another near mid-span, and the third at the wing tip, with straight lines connecting the corresponding coordinates.

By judicious selection of the camber and thickness ratios of the three controlled sections, it becomes possible to obtain spanwise distributions of maximum section lift coefficients similar to that shown in Fig. 7. A comparison of the spanwise distributions of actual and maximum attainable section lift coefficients discloses that the previously postulated requirement of a mid-span stall progressing evenly inboard and outboard is met.

A convenient procedure for the selection of the most appropriate parameters (camber and thickness ratio) for the three controlled sections is based on the fundamental information of the variation of maximum lift and zero-lift angle with camber, thickness ratio, and Reynolds Number for a given airfoil, required for the respectively selected airfoil family.

A preliminary selection of the three controlled airfoil sections is undertaken, mainly on the basis of past experience. The camber and thickness ratios of several intermediate stations are then determined and the variation of $C_{L_{max.}}$ vs. Reynolds Number is plotted for these representative airfoil sections (Fig. 8).* For actual calculations a complete curve of $C_{L_{max.}}$ vs. log (R.N.) must be used (for example, Fig. 28 of reference 3). Assuming the approximate air speed at which the stall is expected, the Reynolds Numbers of the various spanwise stations are computed and plotted on the $C_{L_{max.}}$ vs. Reynolds Number graph. The resulting curve of maximum lift coefficients is then transferred to the graph of $C_{L_{max.}}$ vs. span. If the resulting relation between the $C_{L_{max.}}$ available curve and the span-load distribution is not satisfactory, minor adjustments of the camber, thickness ratios, and the washout will modify the two spanwise distributions until the desired result is obtained.

The variation between maximum lift coefficients and thickness ratio shows a certain peculiarity that can be employed to good advantage. Most airfoil families reach their highest $C_{L_{max.}}$ at a thickness ratio between 12 and 16 per cent. Thickness ratios greater or smaller than the optimum value result in lower maximum lift coefficients. Consequently, if a thickness smaller than optimum is used for the wing tip, where the load is greatly reduced from its peak value, the optimum airfoil thickness can be located at the spanwise station a small distance inboard of the wing tip where the highest load is reached (Fig. 9).

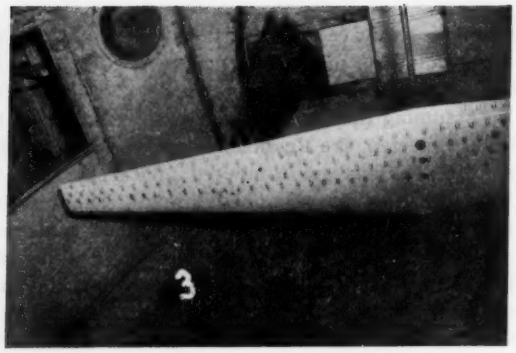
Wind-Tunnel Testing for Stalling Characteristics

Wind-tunnel testing on small-scale models for the prediction of the full-scale stalling characteristics is generally not entirely satisfactory because it is extremely difficult to reproduce the full-scale Reynolds Number without exceeding the full-scale Mach Number. This is particularly disconcerting when testing in small, atmospheric tunnels during the preliminary-design stage of a new-type aircraft, at which phase accurate data for the estimation of the full-scale stalling characteristics are most urgently required. Some assistance, at least, on this perplexing problem can be gained from the $C_{L_{max.}}$ vs. Reynolds Number graph, where model Reynolds Numbers are used instead of full-scale values.

No general rule on the comparative character of the stalling characteristics at model and full-scale can be advanced but it is recommended that a prediction of the model stalling characteristics be made prior to the wind-tunnel test, not only to test the accuracy of the method, but also to uncover the existence of any unforeseen interference factors, such as the individual and combined effect of fuselages, nacelles, and propeller slipstream on the stall characteristics.

The method has been successfully tested on wings with taper ratios up to 4:1 and leading edge sweep-

* Fig. 8 shows only the linear range of the C_L variation with log (Reynolds Number).



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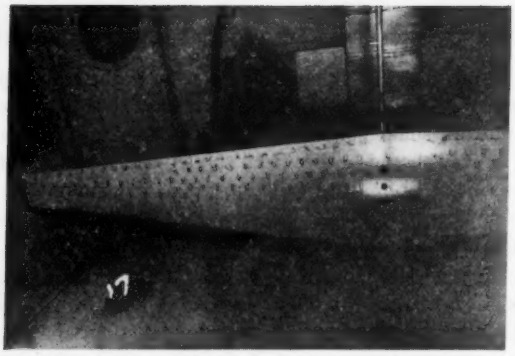

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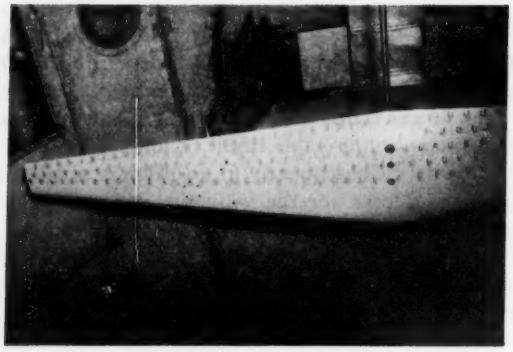

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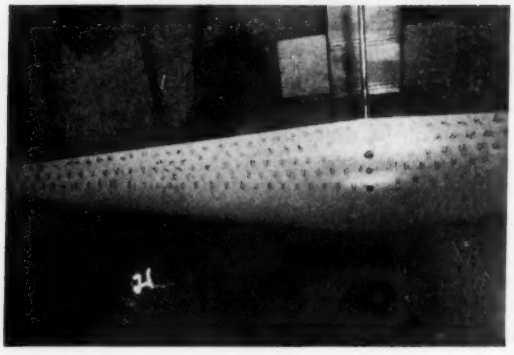

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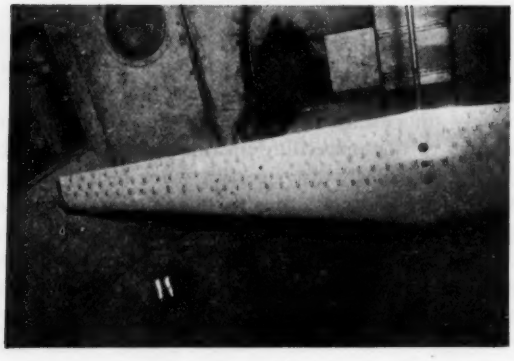

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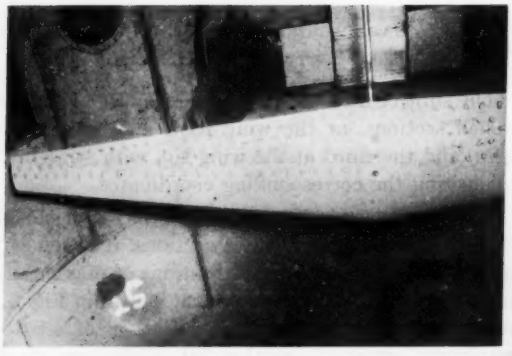

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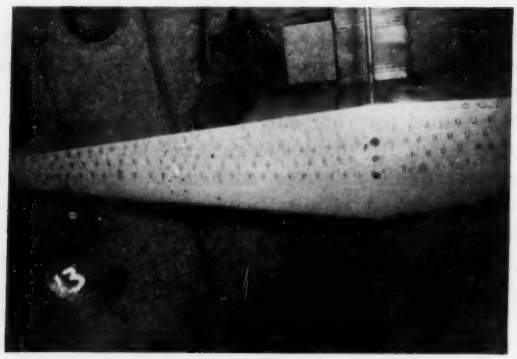

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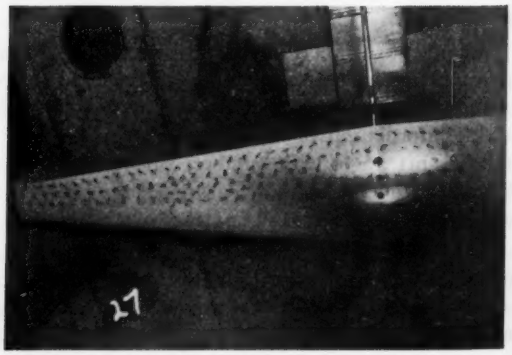

$\alpha = 17°$

FIG. 12.

backs up to 15°. Because of military restrictions the visual demonstration of stall characteristics on a wind-tunnel model must be limited to photographs of a non-confidential research wing with taper ratio 3:1, which is, however, fully representative of wings with higher taper ratios and greater sweep-back. The airfoils used are NACA 2518, 3515, and 4512, respectively.[4] No aerodynamic washout is incorporated. A theoretical comparison of the stalling characteristics of this wing and a wing with straight-line fairing between a 2518 root airfoil and a 4512 tip airfoil (no aerodynamic washout) is shown in Fig. 10. It is of significance that the stall of the "tri-section wing" begins at a wing-lift coefficient of 1.5 against a stalling-lift coefficient of 1.4 in a conventional straight-line faired two-section wing. The lift and pitching-moment data actually obtained at the wind tunnel (not extrapolated for scale effect; Fig. 11) and the photographs of Fig. 12 substantiate the concurrence of estimated and experimentally observed characteristics of the tri-section wing.

Conclusion

The adoption of a third controlled airfoil section near mid-span permits the attainment of any desired stall characteristics by eliminating the localized deep stalls over the outboard panels. A desirable apportionment of spanwise lift distribution at relatively high lifts can be determined for given stall characteristics because a satisfactory stall can be obtained with a smaller spanwise variation of camber.

References

[1] Anderson, Raymond F., *A Comparison of Several Tapered Wings Designed to Avoid Tip Stalling*, N.A.C.A. T.N. No. 713, 1939.

[2] Anderson, Raymond F., *Determination of the Characteristics of Tapered Wings*, N.A.C.A. T.R. No. 572, 1933.

[3] Jacobs, Eastman N., and Sherman, Albert, *Airfoil Section Characteristics As Affected by Variations of the Reynolds Number*, N.A.C.A. T.R. No. 586, 1937.

[4] Jacobs, Eastman N., Ward, Kenneth E., and Pinkerton, Robert W., *The Characteristics of 78 Related Airfoil Sections from Tests in the Variable-Density Wind-Tunnel*, N.A.C.A. T.R. No. 460, November, 1933.

Letters to the Editor

Dear Sir:

For some time our company has used the method of compressibility correction which was derived by Dr. Greene and published in Volume 12, Number 3, July, 1945, of the Journal of Aeronautical Sciences, under the title, "The Attenuation Method for Compressible Flow Systems." We have found that by using this method properly we could obtain close agreement between our calculated incipient shake-boundary and that obtained from flight tests.

To do this we calculate the wing airfoil pressure distributions at several angles of attack and a representative station for zero Mach Number and then calculate these same pressure distributions for a Mach Number such that the peak negative pressure has just reached the line of critical C_p as shown in Fig. 3 of Dr. Greene's article. These pressure distributions are then integrated giving us values of C_{L_m} which can then be plotted against their respective Mach Numbers. This, then, is the incipient shake-boundary.

Experience with the FR-1 airplane showed that this curve substantially represents the incipient shake-boundary except at high lift coefficients where the normal or "zero Mach Number" stall interferes.

R. B. Johnston
Chief Aerodynamicist
Ryan Aeronautical Company

Dear Sir:

This letter presents a shortened workup for the Jones formula used in determining profile drag by the pitot-traverse method. It proposes a method by which profile drag coefficients may be obtained directly from the wake survey data without lengthy calculations.

When the wake has reached a point about 0.7 chord behind the trailing edge of the wing, the wake static pressure, p, is essentially

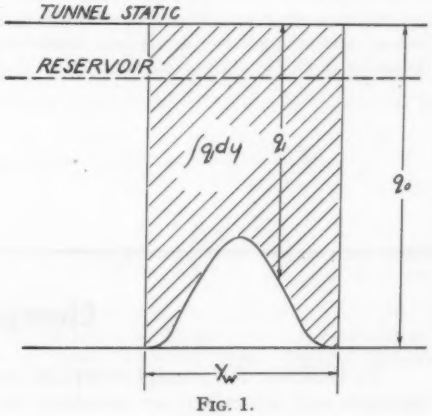

Fig. 1.

equal to the freestream static pressure, p_0, and the Jones equation for drag*

$$c_{d_0} = 2\int \sqrt{\frac{H_1 - p_1}{H_0 - p_0}}\left(1 - \sqrt{\frac{H_1 - p_0}{H_0 - p_0}}\right) d\left(\frac{y}{c}\right) \qquad (1)$$

where H_1 and H_0 are total pressure in the wake and freestream, y is the wake location of the tube reading the total head, and c is the wing chord, reduces to

$$c_{d_0} = 2\int [\sqrt{q_1/q_0} - (q_1/q_0)]d(y/c) \qquad (2)$$

q_1 and q_0 are the dynamic pressures in the wake and freestream, respectively.

Further, at this distance the ratio of q_1/q_2 varies from 0.80 to 1.00. In this range the expression

$$\sqrt{q_1/q_0} = (q_1/2q_0) + 0.5 \qquad (3)$$

* Jones, B. Melville, "Measurement of Profile Drag by the Pitot Traverse Method," R. & M. No. 1688, British ARC, 1936.

is valid within 0.5 per cent. The maximum error of 0.5 per cent occurs only at the wake peak, and the overall error in c_{d_0} becomes of the order of 0.25 per cent.

Substituting Eq. (3) in Eq. (2),

$$c_{d_0} = 2\int\left[\left(\frac{q_1}{2q_0}+0.5\right)-\frac{q_1}{q_0}\right]d\left(\frac{y}{c}\right) = 2\int 0.5\,d\left(\frac{y}{c}\right) - \int\left(\frac{q_1}{q_0}\right)d\left(\frac{y}{c}\right) = \frac{y_w}{c} - \frac{1}{q_0 c}\int q_1 dy \qquad (4)$$

where y_w = wake width.

Eq. (4) has the advantage that it can be integrated directly from the original data. The value y_w can be measured directly on the manometer bank, or whatever method is used to present the pressures, and the value of $\int q_1 dy$ can also be integrated directly if the manometer tube spacing is proportional to the tube spacing on the wake survey rake (Fig. 1).

ALAN POPE
Associate Professor of Aerodynamics
Daniel Guggenheim School of Aeronautics
Georgia School of Technology

Dear Sir:

In the recent paper, "A General Stability Index for Flow Near Plane Boundaries," (JOURNAL OF THE AERONAUTICAL SCIENCES, Vol. 12, No. 4, p. 429, October, 1945), Hunter Rouse develops a stability parameter χ on the basis of dimensional analysis. The arguments underlying the formulation of χ are open to some criticism. Before discussing Professor Rouse's approach, however, it is well to note that a rigorous mathematical theory of laminar instability for plane motion exists today. The stability investigations that were begun by Heisenberg, Tollmien, Schlichting, and others have recently been developed into a general theory by C. C. Lin (*Proc. Nat. Acad. Sci.*, Vol. 30, p. 316, 1944; *Quart. of Appl. Math.*, Vol. 3, pp. 117–142 and 218–234, 1945). Experimental research carried out during the war gave excellent qualitative and quantitative agreement with Lin's theory for boundary-layer flow. Most of these results were unavailable to Professor Rouse at the time of submittal of his paper and are quoted here for completeness rather than as a criticism.

It can be argued, of course, that the highly mathematical stability theory is of little use to the practical engineer and that a simple stability criterion should be developed. Professor Rouse's parameter, however, does not include all the essential quantities which characterize the instability and thus can hardly represent the stability of a given laminar flow pattern. For example, the wave length of the disturbance λ and the curvature of the profile d^2v/dy^2 are two of these important variables. The Couette flow where d^2v/dy^2 vanishes is, for example, theoretically stable. Professor Rouse's stability criterion, however, gives instability for this flow. The application of Professor Rouse's parameter to flow between rotating cylinders is not justified, because here the action of the centrifugal forces that give rise to the three-dimensional Taylor vortexes are of primary importance. The case of rotating cylinders is, in fact, the first flow pattern for which a quantitative stability theory was worked out and found in complete agreement with experimental evidence (G. I. Taylor, *Phil. Trans. A*, Vol. 223, p. 289, 1923). A stability parameter for this case has to include at least the radius of curvature, and this flow pattern is thus not readily comparable with the flow past a flat plate and pipe flow. The problem of laminar instability has proved too complicated to permit the formulation of simple criteria without consideration of the mechanism leading to instability. A simple stability parameter can only be extracted from the general theory. Attempts in this direction can be found in the recent papers by C. C. Lin.

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The Determination of the Stalling Speed and the Maximum Lift Coefficient in Flight

ERNEST W. GRAHAM* AND HAROLD LUSKIN†

Douglas Aircraft Company, Inc.

SUMMARY

An analysis has been made of the errors‡ that may be encountered in conducting stalling speed flight tests. The errors were found to result from four causes: pilot technique, test conditions, airplane configuration, and instrumentation. A tabular breakdown of these items is presented, with an estimate of their approximate order of magnitude and an appraisal of the possibilities of eliminating or correcting the errors. The largest errors resulted from inadvertent normal accelerations (particularly in the case of unusual approaches to the stall, as in level flight), power, and trailing bomb lag. The more important small errors include those due to flight path inclination, angular velocity in pitch, and induced velocity at the bomb. Procedures and corrections were considered in an effort to find means for eliminating the errors or their effects.

As a concluding portion of the study, a procedure** is recommended for the flight determination of the power-off maximum lift coefficient and stalling speed. This procedure specifies the following:

1. A deceleration of no greater than 0.5 m.p.h. per sec.
2. Zero thrust set at not to exceed 10 per cent above the stalling speed.
3. Use of a trailing bomb, with a correction for induced velocity.
4. A correction for flight path inclination. (To be applied to maximum lift coefficient but not to stalling speed.)
5. Use of sensitive accelerometers.
6. Measurement of the stalling speed as that at which uncontrollable pitching takes place or the minimum obtainable in steady controlled flight.

INTRODUCTION

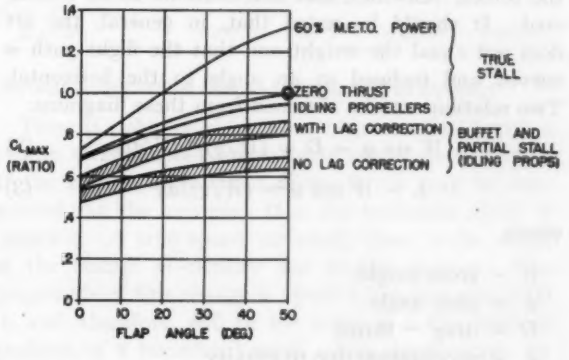
THE MAXIMUM LIFT COEFFICIENT, because of its direct bearing on problems of take-off, landing, and maneuverability, is of great importance to airplane manufacturers, air-line operators, and pilots. Accurate measurement of this quantity in flight is essential to the development of aerodynamic improvements which ultimately increase the operating economy of the airplane.

In addition, the Civil Air Regulations define certain limits to the allowable nontransport airplane power-off stalling speeds†† and rates of climb of transport and nontransport aircraft. The Regulations defining required rates of climb are based on the stalling speed and the stalling speed squared; a 2 per cent error in the stalling speed may affect the pay load by about $3\frac{1}{2}$ and 7 per cent, respectively.

In view of the importance of the maximum lift coefficient and considering the wide variation in values of $C_{L_{max.}}$ which can be measured in flight as illustrated in Fig. 1, it was decided to investigate the errors that may be encountered during the tests for $C_{L_{max.}}$. These errors are classified under four headings: pilot technique, test conditions, airplane configuration, and instrumentation. The effect of any error is to alter the variables in parentheses in the fundamental relation:

$$C_{L_{max.}} = \frac{391L}{SV_{i_{true}}^2} = \frac{391\,W_{recorded}}{SV_{i_{observed}}^2}\left(\frac{W_{true}}{W_{recorded}}\right) \times \left(\frac{L}{W_{true}}\right)\left(\frac{1}{1+\frac{\Delta C_{L_{max.}}}{C_{L_{max.}}}}\right)\left(\frac{V_{i_{observed}}}{V_{i_{true}}}\right) \quad (1)$$

FIGURE 1
EFFECT OF POWER, LAG, AND BUFFETING ON APPARENT $C_{L_{MAX}}$



Received February 1, 1945.

* Assistant Chief of Aerodynamics Section, Santa Monica Engineering Division.

† Aerodynamics Research Engineer, Santa Monica Engineering Division.

‡ Any variations from a standard power-off $C_{L_{max.}}$ (obtained by the procedures recommended here) shall, for simplicity, be termed errors.

** Various methods have been proposed for measuring the stalling speed. Its measurement is especially important to civil aircraft because the performance requirements of the Civil Air Regulations are based largely on the measured stalling speed. At the C.A.A.-Industry meeting on "Flight Testing Technique," held in Hollywood, January 5, 1945, a procedure for stall speed tests was adopted. The procedure suggested here was relaxed in favor of simplicity. A maximum deceleration of 1 m.p.h. per sec. was allowed, no correction for induced velocity at the bomb less than 1 m.p.h. was to be taken into account, and accurate instrumentation was called for, but not necessarily to include accelerometers.

†† The stalling speed, V_s, is defined as the speed a the instant an uncontrollable pitching takes place or the minimum attainable speed in steady controlled flight. Further discussion of this appears later in the report.

FIGURE 2
FORCE VECTOR DIAGRAM FOR GLIDING FLIGHT

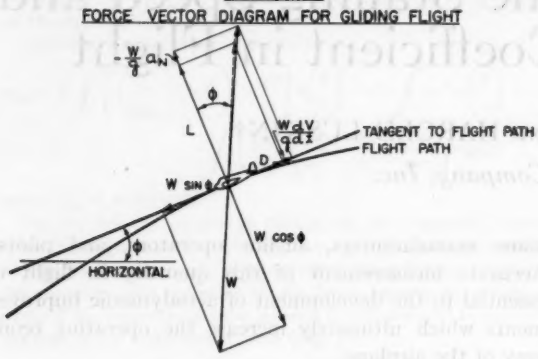
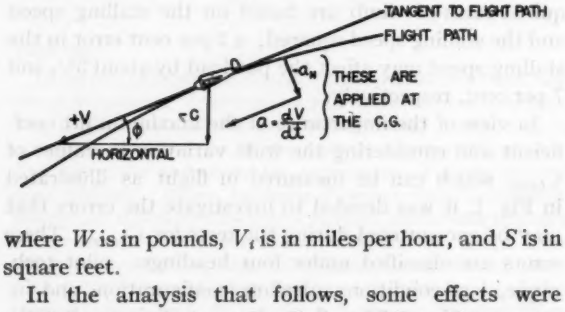


FIGURE 3
VELOCITY AND ACCELERATION VECTOR DIAGRAM FOR GLIDING FLIGHT



where W is in pounds, V_i is in miles per hour, and S is in square feet.

In the analysis that follows, some effects were studied as variations in $C_{L_{max.}}$, while others were studied as variations in $V_{i_{true}}$. The corrections are related by the formula,

$$\% \text{ error in } C_{L_{max.}} = -2 \ (\% \text{ error in } V_i)$$

The portions of this paper which follow are concerned with the evaluation of these effects and with the determination of a procedure for avoiding or correcting the errors.

The Effect of Pilot Technique

The influence of the pilot on the $C_{L_{max.}}$ is the result of his control of the flight path and of the orientation of the airplane to the flight path. Figs. 2 and 3 show the forces, velocities, and accelerations to be considered. It should be noted that, in general, the lift does not equal the weight and that the flight path is curved and inclined at an angle to the horizontal. Two relations can be obtained from these diagrams:

$$W \sin \phi - D = (W/g)/(dV/dt) \quad (2)$$

$$L - W \cos \phi = (W/g)/a_N \quad (3)$$

where

W = gross weight
ϕ = glide angle
D = drag − thrust
g = acceleration due to gravity
V = velocity along flight path
t = time
L = lift
a_N = normal acceleration

These relations will be used to show several interesting facts.

Inclination of Flight Path

For the case of unaccelerated flight, Eqs. (2) and (3) reduce to

$$W \sin \phi = D \quad (4)$$

$$W \cos \phi = L \quad (5)$$

from which is found the lift-weight ratio.

$$\frac{L}{W} = \cos \phi = \sqrt{1 - \sin^2 \phi} = \sqrt{1 - \frac{1}{(W/D)^2}}$$

$$L/W = \sqrt{1 - [1/(L/D)^2]} \quad (6)$$

The magnitude of this factor is 0.98 for L/D of 5. Thus, a 2 per cent discrepancy in $C_{L_{max.}}$ may be caused by the error in the common assumption that the weight equals the lift.

This correction should be made in calculating $C_{L_{max.}}$ from the measured stalling speed. If it is then desired to calculate the power-off stalling speed for this same condition from $C_{L_{max.}}$, it is necessary to apply again the inclination correction but with opposite sign. Thus, if the test conditions are reasonably close to the standard configuration for the test, the stalling speed is not affected by inclination, and this correction can be neglected for obtaining V_s (but not $C_{L_{max.}}$).

Inadvertent Normal Accelerations

This classification is intended to cover accelerations that are not directly predictable from the type of stall approach assumed.

Surprisingly large effects can result from inadvertent normal accelerations. Eq. (3) can be written as

$$(L/W) - \cos \phi = a_N/g \quad (7)$$

Thus for any fixed flight path inclination, each $0.01g$ causes a 1 per cent change in the lift-weight ratio. It is difficult to determine the probable maximum error that might be caused by such accelerations. Certainly $0.01g$ is well below the limit of the pilot's perception, particularly since vibratory accelerations due to buffeting which precede the stall may be of relatively great magnitude under the best of conditions. Data recorded during attempts to perform stalls in level flight have indicated that accelerations of $-0.20g$ can be obtained. A 20 per cent error in $C_{L_{max.}}$ then results if it is assumed that $L = W$. The curvature of the flight path is small for small accelerations and may not be detected by the pilot; an 11-mile radius of curvature corresponds to $0.01g$ at 80 m.p.h. indicated speed

at 10,000-ft. altitude. That this may easily lead to sizable errors can be shown in the following way. Suppose, first, that there was an acceleration, a_N, acting prior to stall. Then the normal acceleration is related to the angular velocity, w, the radius of curvature, R, and the speed.

$$a_N = V^2/R = wV \tag{8}$$

If the acceleration is constant for time t, then the angular movement is

$$wt = (a_N/V)t \tag{9}$$

and the change in altitude, for the case of initially horizontal flight, is

$$\Delta h = (V^2/a_N)\,(1 - \cos wt) \tag{10}$$

and

$$\Delta h = (V^2/a_N)\,[1 - \cos (a_N t/V)] \tag{11}$$

Thus, for an acceleration of 0.10g, and for V_t = 80 m.p.h. at 10,000 ft., the loss in altitude in 2 sec. (or 275 ft. of horizontal distance) is but 6 ft., which is too small to be noted reliably on the altimeter. This clearly shows the possibility that large normal accelerations might remain undetected. Furthermore, the angular changes due to the increase in angle of attack may well tend to hide from the pilot a small curvature in the flight path itself.

Normal Acceleration Due to Constant Rate of Descent

Methods have recently been proposed for conducting stall tests in which the approach to the stall is made at a fixed rate of descent. The reasons for these procedures stem from the fact that actual stalls are most usually encountered in the landing approach or at the extreme, in horizontal motion, as in landing. If a fixed vertical velocity is maintained with a decreasing horizontal velocity, the tangent of the flight-path angle is changing, and the resulting curvature of the flight path always introduces negative normal accelerations.

It is possible to analyze these special stall approaches in a simple fashion, since the condition of fixed rate of descent leads to zero vertical acceleration. Then, the resultant acceleration must be horizontal and have a component *normal* to the flight path, a_N, equal to the resultant multiplied by the sine of the flight-path angle. Since the component, dV/dt, *along* the flight path is equal to the resultant multiplied by the cosine of the flight-path angle,

$$a_N = (dV/dt) \tan \phi \tag{12}$$

After inserting this value in Eq. (3) and eliminating between Eqs. (3) and (2),

$$L - W \cos \phi = (W \sin \phi - D) \tan \phi \tag{13}$$

which reduces to

$$W/L = \sqrt{1 - (C^2/V^2)} - (D/L)(C/V) \tag{14}$$

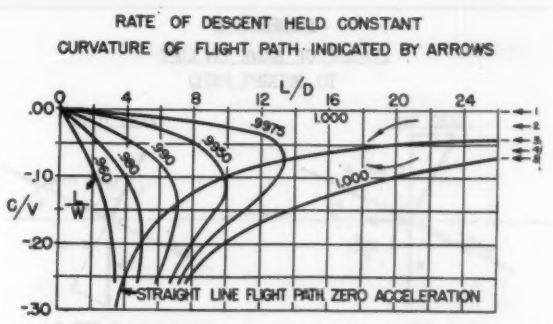


FIG. 4. Lift to weight ratio as a function of L/D and glide angle.

1. Level flight stall approach.
2. V decreasing. a_N negative.
3. V constant. a_N zero.
4. V increasing. a_N positive.
5. V increasing. a_N positive. Effect of a_N exactly balances effect of inclination.

where $-C$ = rate of descent and the other symbols are as before. For the special case of unaccelerated flight, Eq. (14) reduces to Eq. (6). Eq. (14) is plotted on Fig. 4. The corrections, in general, are small and reach a maximum for the unaccelerated stall technique. It should be kept in mind that the errors here are due to both inclination and curvature of the flight path. To indicate the relative magnitudes of the two effects, Table 1 has been prepared. It can be seen on this table that the error due to inclination (depending on the cosine of the flight-path angle) always increases with an increase in glide angle, whereas the error due to curvature is zero in horizontal flight, increases to a maximum negatively as the path steepens, again reduces to zero, then changes sign and increases positively as the path becomes even steeper. The change in sign of the curvature error is associated with a change in path curvature from concave down to concave up.

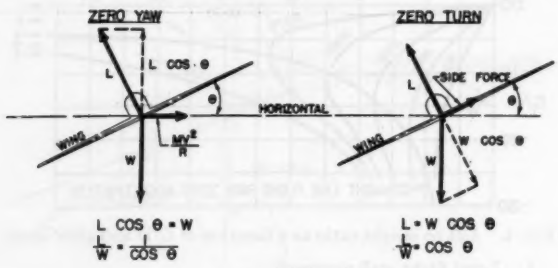
TABLE 1
Per Cent (L/W) Changes Due to Path Inclination and Curvature $(L/D = 6.5)$

C/V	0	− 0.100	− 0.154	− 0.200
Flight path (not to scale)	—	✓	✓	✓
% $\Delta L/W$ due to inclination	0	− 0.50	− 1.16	− 2.02
% $\Delta L/W$ due to curvature	0	− 0.53	− 0.00	+ 0.97
% $\Delta L/W$ total	0	− 1.03	− 1.16	− 1.05
% $L/W = 100 +$ % $\Delta(L/W)$	+100.0	+98.97	+98.84	+98.95

Normal Acceleration Due to Constant V_t in Descent

There is a change of kinetic energy associated with a change of true speed which can occur in several ways. If the stall is approached gradually, it may be considered (at the extreme) that the indicated speed is constant. A true speed variation, then, is the result of the change of density due to the descent. The magnitude of this change is about 1.5 m.p.h. per 1,000 ft. and, therefore, will be the same as a like vertical gradient of a horizontal wind. This deceleration, by

FIGURE 5
EFFECT OF BANK ON LIFT TO WEIGHT RATIO



Newton's law, gives rise to an effective horizontal force, which does work at a rate proportional to the velocity, the work appearing as an increment in rate of descent. Calculations on the order of magnitude of these effects show them to be negligible (see Appendix).

Angle of Bank

Angle of bank introduces a variation of the lift-weight ratio which is a function of the cosine of the angle of bank, θ. For zero yaw, $L/W = 1/\cos\theta$. while for zero turn, $L/W = \cos\theta$, as indicated in Fig. 5. The effect is small (± 0.5 per cent for 6° angle of bank).

Tail Load Required to Produce Angular Acceleration

This effect is small but the equations developed are useful in other investigations. For this reason it is desirable to present the derivation here.

If the stall is approached in any way but with practically no deceleration along the flight path, angular velocity and angular acceleration about the pitching axis may result. A tail load, the magnitude of which will now be computed, is required to maintain the angular acceleration. The effect of c.g. position on this required dynamic tail load is negligible.

Assume the variation of drag coefficient with lift coefficient to be parabolic, with the vertex at the origin. This introduces a negligible change in the polar at high lift coefficients. Also assume that the stall technique is such that constant altitude will be maintained, since this will lead to the greatest angular accelerations. Then from Eq. (2)

$$D = -(W/g)/(dV/dt) \tag{15}$$

and

$$dV/dt = -g/(W/D) \tag{16}$$

or approximately,

$$dV/dt = -g/(L/D) \tag{17}$$

Now, according to the assumption for the drag coefficient,

$$D = C_L^2 qS^2/\pi Eb^2 \tag{18}$$

where $q = {}^1/_2\,\rho V^2$, E = a span efficiency factor, and b = wing span. This can be written

$$D = 2L^2/\pi Eb^2\rho V^2 \tag{19}$$

It is a sufficiently good approximation for this purpose to use

$$C_L = 2W/\rho V^2 S \tag{20}$$

The time derivative of C_L is

$$dC_L/dt = -(2C_L/V)(dV/dt) \tag{21}$$

from which dV/dt is eliminated by Eq. (17)

$$(dC_L/dt) = (2C_L/V)/[g/(L/D)] \tag{22}$$

and then

$$d\alpha/dt = 2(C_D g/V(dC_L/d\alpha) \tag{23}$$

where α is the angle of attack. An additional differentiation results in

$$d^2\alpha/dt^2 = 10\, g^2 C_D/(dC_L/d\alpha)\; V^2(L/D) \tag{24}$$

which is conservative because of the type of drag variation assumed. The tail load, T, associated with this acceleration will be

$$T = W\beta^2\,(d^2\alpha/dt^2)/gl \tag{25}$$

where β = pitching radius of gyration and l = tail length. It is a good approximation to assume $\beta/l = 1/\sqrt{10}$. Therefore, the error in the lift-weight ratio will be

$$\Delta L/W = T/W = l(d^2\alpha/dt^2)/10\, g \tag{26}$$

and, finally,

$$\Delta L/W = gC_D l/(dC_L/d\alpha)V^2\,(L/D) \tag{27}$$

For $C_D = 0.337$, $l = 50$ ft., $dC_L/d\alpha = 4.59$ per rad., $L/D = 6.5$, and $V_i = 80$ m.p.h. at 10,000 ft., $\Delta L/W = 0.10$ per cent, which is small enough to ignore.

Dynamic $C_{Lmax.}$

A further consequence of rotation at the stall is the introduction of the so-called dynamic lift. This phenomenon, proportional to the angular velocity at the stall, is said to have several causes. The first is a variation in local chordwise angle of attack which effectively changes the camber. The other effect is due to the fact that the circulation attains abnormally high values before the boundary layer can develop and separate.

The effective camber change can be taken into account, using the formulas of reference 1, by assuming the complete lift curve to be uniformly affected. Then the $\Delta C_{L_{max.}}$ will be

$$\Delta C_{L_{max.}} = 2\lambda\,(c/V)\,(d\alpha/dt) \tag{28}$$

where λ is a constant depending on aspect ratio and c.g. location, with a value of about 1.5 for A.R. of 8 to 10 and c.g. positions of 15 to 20 per cent M.A.C. aft of the

leading edge. c is the mean aerodynamic chord. The effect on lift is

$$\Delta L = 2\lambda\,(c/V)\,(d\alpha/dt)\,(\rho V^2/2)S$$

$$\Delta L = \lambda c\rho\; VS\,(d\alpha/dt) \qquad (29)$$

but $d\alpha/dt$ is known from Eq. (23)

$$\Delta L = \lambda c\rho VS\,[2C_D\; g/V\,(dC_L/d\alpha)]$$

$$\Delta L/W = 2g\lambda\rho S\; C_D\; c/W\,(dC_L/d\alpha) \qquad (30)$$

and, finally,

$$\Delta L/W = [2\lambda\; C_D/(dC_L/d\alpha)]\;[g\rho/(W/cS)] \qquad (31)$$

The last term enclosed in brackets is proportional to the ratio of the specific weights of air and the airplane. For a wing loading of 40 lbs. per sq.ft., a mean aerodynamic chord of 14 ft., and an altitude of 10,000 ft., this term has a value of 0.0198. Assuming further that $C_D =$ 0.337 and that $dC_L/d\alpha = 0.08 \times 57.3 = 4.59$ per rad., the magnitude of $\Delta L/W$ is 0.44 per cent.

The change in $C_{L_{max.}}$ due to time lag before the separation takes place is not the subject of any simple analysis. Some model tests are described in reference 2, where the data show that

$$\Delta C_{L_{max.}} = K(c/V)\,(d\alpha/dt) \qquad (32)$$

The form of this expression is similar to that of Eq. (28). It should be remembered that Eq. (32), being derived from actual experiments, includes the effects given by Eq. (28).

A convenient expression for the dynamic $C_{L_{max.}}$ can be obtained by combining Eq. (32) with Eq. (21).

$$\Delta C_{L_{max.}} = K(c/V)(d\alpha/dt)\,(dC_L/dC_L)$$

$$\Delta C_{L_{max.}} = -2K\,\frac{c}{V^2}\,\frac{d\alpha}{dC_L}\,\frac{dV}{dt} \qquad (33)$$

This expression relates the dynamic maximum lift increment to the deceleration. An alternate relation is desirable for the level flight stall, since the rate of deceleration for this case is a function only of the lift-drag ratio for the airplane.

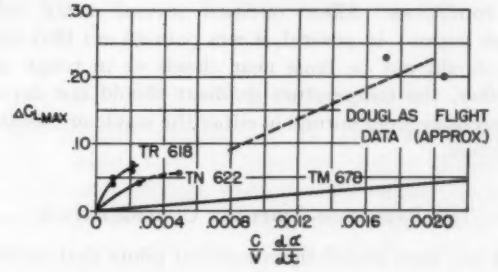
$$\frac{\Delta C_{L_{max.}}}{C_{L_{max.}}} = 2K\,\frac{c}{V^2}\,\frac{d\alpha}{dC_L}\,\frac{g}{(L/D)} \qquad (34)$$

Examination of the data on dynamic $C_{L_{max.}}$ indicates great discrepancies in values of K. The wind-tunnel data of Kramer[2] have $K = 20.6$, while the N.A.C.A. data[3] indicate a K of 20 times this value. Information from other sources shows values of K intermediate to those above. Fig. 6 shows comparative data on dynamic $C_{L_{max.}}$ plotted against angle of rotation per chord length traveled. The average of the data available would lead to $K \doteq 6 \times 20.6$. Inserting this value in Eq. (33), there can be obtained the dV/dt corresponding to any desired accuracy in $C_{L_{max.}}$.

FIGURE 6

$C_{L_{MAX}}$ INCREMENT DUE TO ROTATION IN PITCH

c = WING CHORD, FT.
V = AIRPLANE SPEED, F.P.S.
α = ANGLE OF ATTACK, RADIANS
t = TIME, SECONDS



$$\frac{dV}{dt} = -\frac{V^2}{c}\,\frac{(dC_L/d\alpha)}{250}\,\frac{\Delta C_{L_{max.}}}{C_{L_{max.}}} \qquad (35)$$

For $\Delta C_{L_{max.}}/C_{L_{max.}} = 0.01$, $V_i = 80$ m.p.h. at 10,000 ft., $c = 15$ ft., and $dC_L/d\alpha = 0.08$ per degree, dV/dt can be calculated to be -0.155 m.p.h. per sec. A commonly used figure for deceleration is 0.5 m.p.h. per sec. This corresponds to a 3 per cent increase in $C_{L_{max.}}$.

Variation in Static $C_{L_{max.}}$

There are certain flight attitudes that lead to variations in the static $C_{L_{max.}}$. A rate of roll, for example. alters the lift distribution by introducing an angle of attack increment that is not uniform along the span, Yaw can also alter the lift distribution. It is difficult to say what the magnitude of these variations will be during the approach to the stall because of the great complexity of the problem. The errors can be avoided merely by requiring the pilot to fly with approximately zero roll and yaw.

The Effect of Test Conditions

The classification, test conditions, is taken to include Reynolds Number, Mach Number, wind variations, and air turbulence. No attempt has been made to estimate the magnitude of these effects, since errors due to them can and should be avoided.

Reynolds Number

It seems reasonable to expect that no significant discrepancies due to Reynolds Number can arise if the test gross weight is close to the landing weight and if the altitude for all tests is standardized.

Mach Number

Mach Number will produce insignificant variations in $C_{L_{max.}}$ so long as the stall is made with almost no normal acceleration.

Atmospheric Conditions

The only way to safely avoid errors due to turbulence and wind gradients is to perform tests in stable, smooth air. The most obvious instrument for detecting turbulence is the seat-of-the-pants accelerometer. Certain scientific rules can be applied, however, to check air conditions. Allen[5] outlined several useful rules in his paper. In general, it was pointed out that testing should not be done near clouds or in rough air. Further, the temperature gradient should not depart appreciably from normal in either the stable or unstable direction.

The Effect of Airplane Configuration

It has been stated by competent pilots that stalling speeds can be accurately duplicated on any given airplane at any given time. On different airplanes of the same model, the duplication is said to be poor. It is felt that the reason for these discrepancies lies partly in differences of configuration. Among the items to be considered are power and propeller thrust, center of gravity position, flap position, gear position, leading edge roughness, leaks, aileron trail, cooling flap positions, gross weight, and the like.

Power and Propeller Thrust

Power and propeller thrust can produce tremendous changes in the stalling speed. The mechanism by which this is done is complicated and is associated with changes in velocity and angle of flow in the slipstream, turbulence, change in lift distribution, as well as vertical components of thrust. It has been conventional to idle the propellers during stall tests. In such a case a reduction in lift may occur, due to the decreased slipstream velocity, which can be estimated by using the momentum theory. The fractional increment in lift coefficient will be

$$\frac{\Delta C_{L_{TP}}}{C_{L_T}} = \frac{C_{L_i}}{C_{L_T}} \frac{S_i}{S} \frac{q_s - q_0}{q_0} K \qquad (36)$$

where

$\Delta C_{L_{TP}}$ = increment in total lift coefficient due to power
C_{L_T} = total lift coefficient without power ($T_C = 0$)
C_{L_i} = average C_L for wing in slipstream power off ($T_C = 0$)
S_i = wing area in slipstream
S = total wing area
q_s = dynamic pressure in slipstream
q_0 = dynamic pressure in free stream
K = effectiveness of slipstream q compared to q_0.

From the momentum theory,

$$(q_s - q_0)/q_0 = (8/\pi)T_C \qquad (37)$$

where

$T_C = T/\rho V^2 D^2$
T = propeller thrust
ρ = air density
V = forward speed
D = propeller diameter

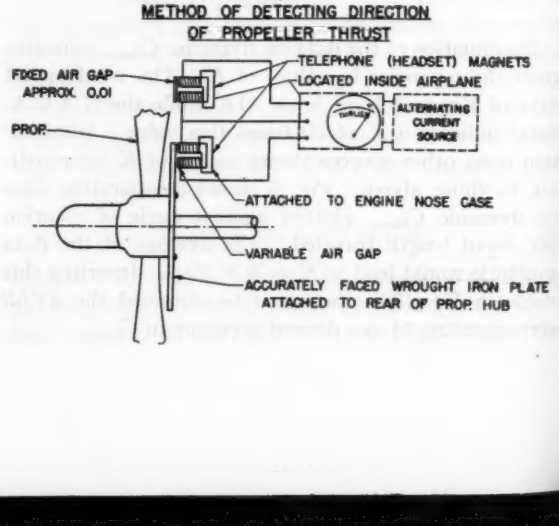
Assuming $T_C = -0.071$, which is representative of high idling negative thrust, $K = 0.5$, to allow for the finite extent of the slipstream and the partial contraction of the slipstream at the wing, $S_i/S = 0.667$ and $C_{L_i}/C_{L_T} = 1.0$, the lift increment is calculated to be $\Delta C_{L_{TP}}/C_{L_T} = -6$ per cent. The lift increments due to turbulence and angle of attack change are not so easily computed. Flight data on one four-engine airplane showed $\Delta C_{L_{TP}}/C_{L_T}$ of -5 per cent.

With gains of such magnitude to be obtained, there may be some inclination to adjust the thrust to positive values, with the justification that pilots normally use some positive thrust even during the landing approach. It would seem reasonable to limit this trend at zero thrust, requiring that zero thrust be obtained in the stall approach slightly above the stalling speed (say at approximately 1.1 V_s). An additional advantage of zero thrust is that all variations of the thrust with temperature, altitude, low pitch setting, oil temperature, idle spring setting, etc., will not affect the stalling speed. In other words, the stalling speed obtained is more nearly a basic aerodynamic number that can be reproduced and checked against other data from various sources and analyses. Simple instruments have been developed to indicate zero thrust from the fore-and-aft motion of the propeller on its shaft. Fig. 7 shows an electrical device that has been satisfactorily flight-tested.

Wing Flap Angle

The variation of $C_{L_{max.}}$ with the position of the wing flaps will depend on the flap design. For one four-engine Douglas airplane, the value of $C_{L_{max.}}$ changes by 0.5 per cent per degree of flap deflection. This is rather a large variation, and it would seem that care should be exercised in selecting and then measuring the flap angle.

FIGURE 7
METHOD OF DETECTING DIRECTION OF PROPELLER THRUST


FIXED AIR GAP APPROX. 0.01
PROP.
TELEPHONE (HEADSET) LOCATED INSIDE AIRPLANE
MAGNETS
ALTERNATING CURRENT SOURCE
ATTACHED TO ENGINE NOSE CASE
VARIABLE AIR GAP
ACCURATELY FACED WROUGHT IRON PLATE ATTACHED TO REAR OF PROP. HUB

Landing-Gear Position

The landing-gear position can affect the stalling speed in several ways. First, if there is a change in trim associated with gear position, the balancing tail load contributes to the lift (positively or negatively). Second, the gear drag appreciably alters L/D, which in turn alters the L/W value. Third, the center of gravity may shift with gear movement, resulting in a change in tail load. Finally, there may be a lift force acting on the gear and gear doors. Corrections for these effects are not ordinarily needed, since the present regulations call for performing the stall tests with the landing gear in specified positions. The magnitude of the effect of the extended gear has been found in flight on certain airplanes to be -2 per cent of the $C_{L_{max.}}$

Gross Weight

Any error in the determination of the gross weight is reflected directly as an error in maximum pay load. To have precise knowledge of the weight, it is vital to know: (a) the basic weight, as weighed; (b) the variation of basic weight due to any and all causes; (c) the weight of pay load, including crew, fuel, oil, and flight equipment; and (d) fuel consumption in flight.

The task of the weight engineer can be seen to be difficult, calling for an almost fanatical attention to detail.

Center of Gravity Position

The c.g. position is usually at the forward limit for stall tests. If it is not, a tail load arises (assuming adequate control) of magnitude

$$\Delta L/W = (x/c)/(l/c) \qquad (38)$$

in which x/c = shift in c.g. position, per cent M.A.C. and

l/c = tail length, per cent M.A.C.

For a stall performed at rear c.g., which is assumed to be 20 per cent M.A.C. aft of the forward c.g., and for $l/c = 2.5$, $\Delta L/W = 8$ per cent. Thus, c.g. position may have a large effect on the apparent maximum lift coefficient.

If adequate control is not available for stalling the airplane at the forward c.g. position, the maximum attainable lift coefficient is seriously reduced. For example, an airplane having just sufficient control to reach $C_{L_{max.}}$ with the c.g. at 20 per cent M.A.C. might attain only 90 per cent of this value with the c.g. at 15 per cent M.A.C.

Miscellaneous Configuration Changes

A great many physical characteristics of the airplane configuration have at one time or another led to erroneous maximum lift coefficients. Among these can be mentioned leading edge roughnesses, leaks, aileron trail, cooling flap positions, fillets, vibration, out of contour surfaces, and the like. Any one of these items can, in the extreme, affect $C_{L_{max.}}$ by at least 0.1. Most of the listed items are built into the airplane and are beyond change, but careful flight engineers can check for these things with some chance for improvement in $C_{L_{max.}}$ on both test and production airplanes. No analytical corrections can here be attempted.

The Effect of Instrumentation

The primary quantity to be determined during stalling speed tests is velocity. Other important measurements include normal acceleration, propeller thrust, c.g. position, fuel quantity, angles of pitch, yaw, and roll, elevator angle, altitude, and time. Photographic recording (or the equivalent) of such data is considered to be necessary for obtaining accurate results. A complete flight engineer's log and split-second time coordination of data are also of great importance. The following discussion is restricted to instrumentation errors that have been a primary source of difficulty in previous flight testing.

Accuracy in speed determination is of extreme importance. The methods for obtaining the speed indication are numerous and each has its advantages. Among these methods are the trailing static bomb, the trailing pitot and static bomb, the trailing log, the mast mounted pitot and static tube, and the pacing airplane. The use of the trailing static bomb will be assumed here, not because it is necessarily the best device but because its characteristics are known to a good degree of accuracy.

Lag Due to Rate of Descent

The lag of a pitot static system is given by the following formula:[*]

$$\Delta V_i = 16(\lambda_S - \lambda_P)\frac{1}{V_i}\frac{dh}{dt} + \lambda_P \frac{dV_i}{dt}\frac{P_0}{P_s} \qquad (39)^*$$

where

ΔV_i	=	correction to indicated speed, m.p.h.
P_S	=	static pressure at altitude
P_0	=	standard static pressure at sea level
V_i	=	indicated air speed, m.p.h.
dh/dt	=	time rate of change of altitude, ft. per sec.
dV_i/dt	=	time rate of change of V_i, m.p.h. per sec.
λ_P	=	pitot lag factor, sec., at standard sea level conditions
λ_S	=	static lag factor, sec. at standard sea level conditions

The lag factor is the time for a given pressure difference to change to $1/e$ times the initial value. In general, λ_S and λ_P are different. It is possible, however, to make $\lambda_S = \lambda_P$ preferably by reducing restrictions

* N.A.C.A. T.N. No. 593, page 15.

in the static line or by increasing the volume of the pitot system or both. (Photographic data recording is vital to this process.) If this is done, the lag due to rate of descent will be zero. One bomb in use at Douglas some time ago had a 10 m.p.h. lag correction at the stall. This shows the great necessity for eliminating lag. λ_s was reduced from 2.8 to 0.38 sec. by changing the tubing size from $^3/_{16}$ to $^3/_8$ in. inside diameter, and the lag due to descent was then reduced to zero by balancing lag factors. This bomb system had a $^3/_{32}$-in. stranded steel cable inside the rubber tubing.

Lag Due to High Deceleration

The procedure just outlined effectively eliminates lag errors due to rate of descent, the first term in Eq. (39), but results in an increase of lag due to rate of deceleration, the last term in Eq. (39). This increase for the last case used above is

$$\Delta V_i = 0.36\ (dV_i/dt)\ P_0/P_s \qquad (40)$$

Then for $dV/dt = 0.5$ m.p.h. per sec. at 10,000 ft., $dV_i/dt = 0.43$, and ΔV is 0.22 m.p.h. above the true V_i, which is usually satisfactorily small. For level flight stalls, however, the errors can be about six times greater than this if the system is balanced for descent, but, if the volume in the pitot system has not been artificially increased, the error should be negligible.

A trailing bomb of new design, featuring electrical transmission of air-speed data, will soon be made available. An advantage of this new design is its elimination of long pressure lines which ends the possibilities of leaks. Leak tests before and after flight are essential with present bombs. A further advantage is the elimination of large lag errors, although the measurement of pitot and static pressure at the bomb introduces several other errors. These include the errors introduced by the vertical gradient of the horizontal wind and the density difference between the airplane and the bomb. Further, oscillations of the bomb relative to the airplane will introduce a varying error, which does not exist on the static bomb. It is considered necessary to calibrate either type of bomb in the wind tunnel and over a speed course.

Induced Velocity Field

The trailing bomb, as well as other devices measuring air speed, measures a speed different from the airplane speed because of the induced velocity field of the wing. Fig. 8 shows the field for a theoretical horseshoe vortex of constant circulation. Checks made for a typical airplane showed the effects of nonuniform spanwise lift distribution to be negligible. The location of the bomb in the field is found using Fig. 9. A flight check of these data on bomb location is shown in Fig. 10. Investigation of several cases has indicated the maximum error in $C_{L_{max}}$ due to induced velocity field to be about 2 per cent. Such errors can be easily and accurately corrected by use of Figs. 8 and 9.

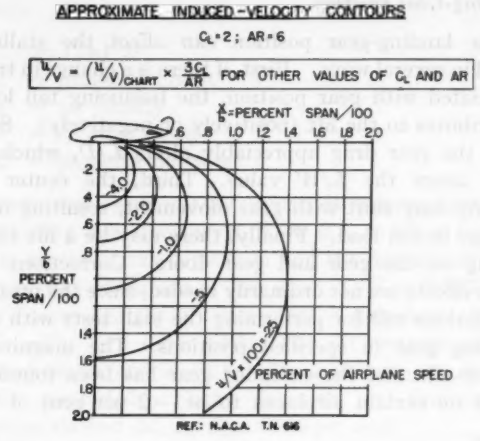


FIG. 8.[7]

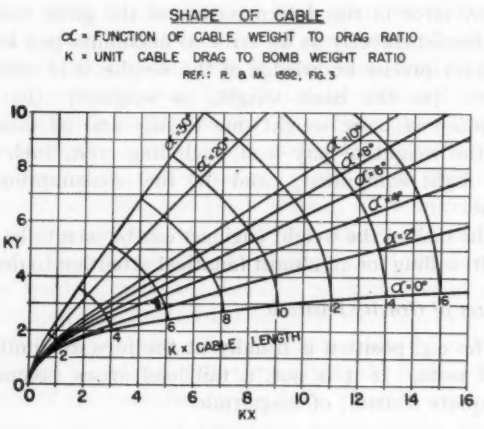

SHAPE OF CABLE
αC = FUNCTION OF CABLE WEIGHT TO DRAG RATIO
K = UNIT CABLE DRAG TO BOMB WEIGHT RATIO
REF.: R. & M. 1592, FIG. 3
$\alpha = 30°$
$\alpha = 22.5°$
$\alpha = 10°$
$\alpha = 7.5°$
$\alpha = 5°$
$\alpha = 2.5°$
$\alpha = 0°$
KY
K x CABLE LENGTH
KX
0 2 4 6 8 10 12 14 16

FIG. 9.[8]

Instrument Calibration and Reading Accuracy

The air-speed indicators in general use today can be the source of several errors. First, the readability limit of about ±0.5 m.p.h. leads to a weight or $C_{L_{max}}$ error of ±1.2 per cent. Second, the mechanical hysteresis may introduce appreciable errors. This can be largely eliminated by instrument selection through proper calibration methods. The instrument hysteresis is found by comparing calibrations made with ever increasing pressure and then with ever decreasing pressure. Differences of 5 m.p.h. have been found on certain occasions, while, on the other hand, a good instrument has a hysteresis of only several tenths of 1 m.p.h.

Solar Heating of the Bomb Cable

The effect of solar radiation on the measured air speed has been investigated and found to be negligible. The cause of this phenomenon is the difference in density of the sun-heated air in the cable and the unwarmed ambient air. Thus, the weight of the air in the cable is lighter than the ambient air, with the result that the

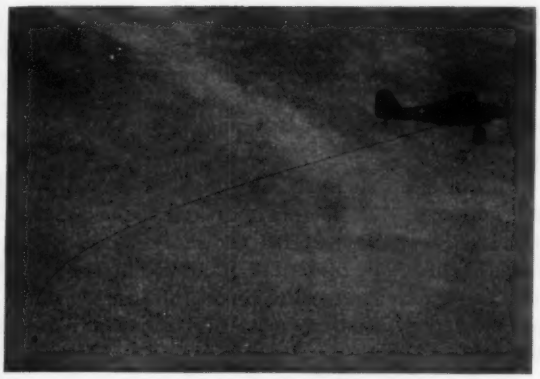
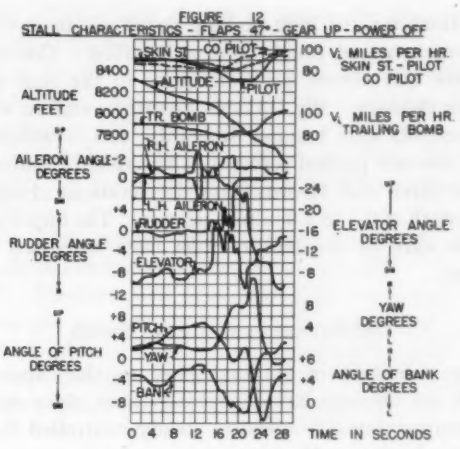
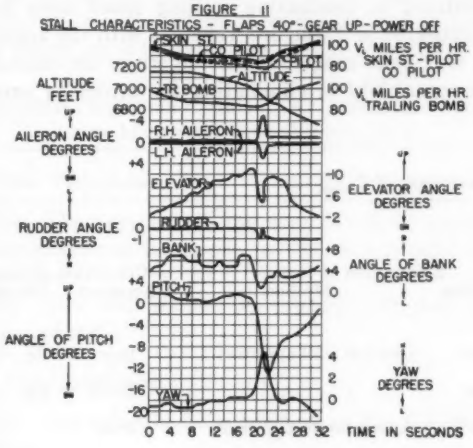
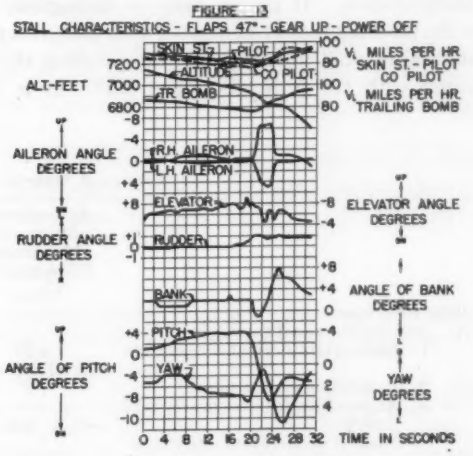


FIG. 10. Douglas trailing bomb being towed at 105 m.p.h. (indicated speed). • Indicates position of bomb according to Fig. 9.







static pressure at the air-speed indicator is in error (see Appendix).

THE MEASURED STALLING SPEED IN FLIGHT RECORDS

After considering the errors involved in the measurement of the maximum lift coefficient it is desirable to investigate flight records and give further consideration to the definition of stalling speed.

Discussion of Stall Records

Some of the difficulties encountered are indicated in Figs. 11–13, which show records of actual stall tests. One of the primary difficulties in these tests is the absence of accelerometer readings. For example, the average curvature of the flight path that appears in the region where pitching first occurs corresponds to a negative acceleration of approximately 0.3*g* maintained for 3 sec., so, when accelerometer readings are not available, the other data recorded are of questionable value. Accelerations appearing in the approach to the stall have been discussed previously.

All three stall histories were obtained during the same flight. The configurations are identical with the exception of small differences in flap angle. The data show angles of bank, pitch, and yaw, positions of elevator, rudder, and ailerons, altitude, and air speed. The four air-speed indications were obtained from the trailing bomb, pitot static tubes on each side of the cockpit, and static holes in the side of the fuselage. The differences in air-speed readings show the importance of knowing all the air-speed corrections. The greatest speed error shown here at time of stall is 7 m.p.h. In each case the stall is associated with a large increase in the rate of descent, a speed nearly the minimum speed, and a definite pitching, yawing, and banking. The stall speeds attained are 83, 82, and 77 m.p.h., figures with large enough variation to be worth investigating. From Fig. 11, it is evident that the elevator angle was reduced sharply at the time of the supposed stall. The high stall speed for this case would seem to be due to the fact that the pilot pushed the stick forward, probably during the period of buffeting. This was the cause of the pitching, while the unsymmetrical unstalling of the small sections of stalled area may have caused the slight bank and yaw. In Fig. 12, this trouble is avoided, and the elevator angle was increased at the time of the stall. The likely ex-

planations for the high stalling speed are inadvertent accelerations and rough control handling. Curves of elevator and aileron positions prior to the stall show sudden changes. Fig. 13 demonstrates a stall in which the controls were handled smoothly, and in which the stick was not pushed forward at the time of buffeting.

The three stall records show decelerations along the flight path of 0.5 to 1.0 m.p.h. per sec. The importance of the rate of deceleration has been discussed previously.

Definition of Stalling Speed

The definition of stalling speed as the "speed at which an uncontrollable pitching takes place or the minimum attainable speed in steady controlled flight" seems adequate. However, some care is required in the interpretation. It is necessary to distinguish between the pitching that is associated with pilot reaction to buffeting or stall warning, and the pitching that is associated with the fully developed stall. It is also necessary to decide whether the more critical of the two alternative speeds must always be designated as the stalling speed or whether the "minimum attainable speed in steady controlled flight" applies only in cases where there is insufficient elevator control to develop a full stall.

It seems desirable in all cases that the stalling speed determined should be one that can be closely approached (say within 2 or 3 per cent) in steady controlled flight. The procedure and corrections outlined in this report are intended to ensure this index of safety, so that special measurements of this nature should normally be unnecessary.

Results of the Study

The results of the foregoing analysis of the errors encountered in conducting stalling speed tests have been collected in Table 2. Together with the approximate maximum magnitude of the errors are estimates of the possibility of eliminating the errors by proper

Table 2
Sources of Error in Flight Determination of $C_{L_{max}}$

		Approximate Maximum Magnitude (% Apparent $C_{L_{max}}$)	Elimination of Error: Possibility	Elimination of Error: Method	Correction of Error: Accuracy	Correction of Error: Method
I.	Pilot technique					
	A. Net lift not equal to weight					
	1. Inadvertent normal acceleration	±20 (Level approach)	Good	Gradual stall approach	Fair	Eq. (7)
	2. Inclination of flight path	+2	Poor	.....	Good	Fig. 4
	3. Angle of bank (zero yaw or zero turn)	±0.5 (for 6°)	Good	Approximate zero bank	Poor	...
	4. Normal acceleration due to constant rate of descent	+1	Good	Gradual stall approach	Good	Fig. 4
	5. Normal acceleration due to constant V_i in descent	Negligible	....	.....	...	...
	B. Variation in tail loads					
	1. Resulting from tail load required to produce angular acceleration	Negligible	....	.....	...	...
	C. Variation in static $C_{L_{max}}$					
	1. Resulting from angle of yaw	?	Good	Approximate zero yaw	Poor	...
	2. Resulting from rate of roll	?	Good	Approximate zero rate of roll	Poor	...
	D. Introduction of dynamic $C_{L_{max}}$					
	1. Resulting from angular velocity in pitch	(3 % per 0.5 m.p.h. per sec.)	Good	Gradual stall approach	Poor	...
II.	Test conditions					
	A. Reynolds number	Small	Good	Test at low altitude	Fair	...
	B. Mach number (unaccelerated flight)	Small	Good	Near standard weight	Fair	...
	C. Wind gradients, turbulence, etc.	?	Good	Avoid	Poor	...
III.	Airplane configuration					
	A. Power and propeller thrust (from zero thrust)	+50, −5	Good	Use zero thrust	Fair	...
	B. Flap position	±0.5/deg.	Good	Test with required position	Fair	...
	C. Gear position	2%	Good	.....	Poor	...
	D. Gross weight (difference between actual and recorded)	±1	Good	Check	Poor	...
	E. Center of gravity location (from critical c.g.)	+8	Good	Test at critical c.g.	Good	Eq. (38)
	F. Leading edge roughness, leaks, aileron trail, etc., each	±5	Good	Careful check	Poor	...
IV.	Instrumentation (trailing bomb static system)					
	A. Lag at high rate of descent	−25	Good	Balance system	Good	Eq. (39)
	B. Lag at high deceleration	−15	Good	Gradual stall approach	Good	Eq. (39)
	C. Induced velocity field	+4	Poor	.....	Good	Figs. 8, 9
	D. Instrument calibration (and reading accuracy)	±2	Good	Careful check	Poor	...

procedure or eliminating their effect by corrections.

Significant errors in $C_{L_{max}}$ can be caused by power, trailing bomb lag, inadvertent normal accelerations, c.g. location, angular velocity in pitch, induced velocity field, inclination of the flight path, and instrument calibration. All of these errors can be adequately handled by proper procedures and corrections.

Proposed Procedure for Conducting Stalling Speed Flight Tests

By considering Table 2 and the present regulations (CAR 04.7511-T[9]), it is possible to set up a procedure for performing stalls so that all the errors considered here can be made satisfactorily small. The use of such a procedure should obtain significant measurements of stalling speed and maximum lift coefficient which can be useful where these values need be known. The procedure is presented in outline form below.

Proposed Procedure for Flight Determination of the Maximum Lift Coefficient

I. Pilot Technique

(A) The approach to the stall should be gradual.* A deceleration of 0.5 m.p.h. per sec. is suggested as an upper limit.

(B) The stall should be entered with zero yaw, bank, and roll.

II. Test Conditions

(A) Tests should be conducted at the lowest safe altitude.

(B) Gross weight should be near the estimated landing weight.

(C) Tests should not be conducted in or near clouds or in turbulent, gusty air. Temperature lapse rates appreciably different than standard should be avoided.

III. Airplane Configuration

(A) The propellers should be idled at zero thrust slightly above the stalling speed (say up to 10 per cent).

(B) Roughness, aileron trail, leaks, etc., should be the same as expected on production airplanes.

(C) The center of gravity should be in the most unfavorable position.

(D) The flaps should be accurately set in the required position.

IV. Instrumentation

(A) The trailing bomb system should be used. Lag should be eliminated by matching pitot and static lag factors at the stalling speed. Leak checks should be made before and after every flight.

(B) Corrections should be made for the induced velocity at the bomb.

(C) Air-speed indicators should be calibrated in such a fashion that instruments with excessive mechanical hysteresis can be eliminated.

(D) Accelerometers (sensitive to 0.01g) should be installed to provide knowledge of the normal acceleration.

(E) The airplane gross weight at the time of the stall should be known within ± 0.5 per cent.

(F) Photographic data recording, at a rate of 2 frames per sec. or faster, should be used to give time histories of the tests. Instrumentation could well include velocity; normal acceleration; elevator position; pitch, bank, and yaw angles; altitude; and time.

V. The Measured Stalling Speed

(A) The stalling speed should be defined as the speed at which an uncontrollable pitching takes place or the minimum attainable speed in steady controlled flight.

(B) Care should be taken to distinguish between the preliminary pitching and pilot reaction (associated with buffeting and stall warning) and the true stall.

* Corrections for flight path inclination can be easily and accurately made if it is desired to know true $C_{L_{max}}$.

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Appendix

Normal Acceleration Due to Constant V_i in Descent

By Newton's Law, the work made available because of deceleration appears as a change in rate of descent. Thus

$$(W/g)(dV/dt)V = W\Delta C \tag{41}$$

Now

$$dV/dt = (dV/dh)C \tag{42}$$

where h is the geometrical altitude. Eq. (41) can be reduced to

$$\Delta C/C = -(1/2g)/[d(V^2)/dh] \tag{43}$$

By definition,

$$V^2 = V_i^2/\sigma \tag{44}$$

where V_i is the indicated air speed and σ is the ratio of the air density to standard sea-level air density. Substituting for V^2 in Eq. (43) and holding V_i constant reduces the equation to

$$\Delta C/C = (V_i^2/2g)(d\sigma/\sigma^2 dh) \tag{45}$$

To evaluate $d\sigma/\sigma^2 dh$, the gas law is used.

$$p = \rho gRT \tag{46}$$

where p = air pressure, ρ = air density, R = universal gas constant, and T = absolute air temperature. The gas law can be differentiated, leading to

$$dp/p = (dT/T) + (d\sigma/\sigma) \tag{47}$$

The evaluation of $d\sigma/\sigma^2 dh$ is now possible.

$$\frac{d\sigma}{\sigma^2 dh} = \frac{1}{\sigma dh}\left(\frac{dp}{p} - \frac{dT}{T}\right) = \frac{1}{\sigma T}\left(\frac{T}{p}\frac{dp}{dh} - \frac{dT}{dh}\right) \tag{48}$$

The equilibrium of the atmosphere is defined by the following expression:

$$dp/dh = -\rho g \tag{49}$$

Also, the temperature gradient will be called $-\theta$. Then,

$$d\sigma/\sigma^2 dh = (-\rho_0 g/p)\,(1 - R\theta) \tag{50}$$

Eqs. (45) and (50) can now be combined to give the expression for the percentage effect of the change of true speed on climbing rate.*

$$\Delta C/C = (-V_i^2\rho_0/2p)\,(1 - R\theta) \tag{51}$$

Now, remembering that $\Delta C/C = \Delta C_i/C_i$, the inclination of the flight path is

$$\phi = \arcsin\,[-(C_i + \Delta C_i)/V_i] \tag{52}$$

which can be written as

$$\phi = \arcsin\left[-\frac{C_i}{V_i}\left(1 - \frac{V_i^2\rho_0[1 - R\theta]}{2p}\right)\right] \tag{53}$$

If the glide path is curved, there will be associated with it Ω, an angular velocity, in radians per second.

$$\Omega = \frac{d\phi}{dt} = \frac{-\left(\dfrac{C_i\,V_i^2\rho_0(1 - R\theta)dp}{V_i 2p^2 dt}\right)}{\sqrt{1 - \dfrac{C_i^2}{V_i^2}\left[1 - \dfrac{V_i^2\rho_0(1 - R\theta)}{2p}\right]}} \tag{54}$$

Eq. (49) can be used here, and approximately,

$$\Omega \doteq \frac{\rho_0^{3/4} g\,(1 - R\theta)C_i^2\,V_i\rho^{1/2}}{2p^2} \tag{55}$$

this angular velocity gives rise to a normal acceleration.

$$a_N = \Omega V \tag{56}$$

Using Eq. (2), an expression for L/W can be written

$$L/W = (V\Omega/g) + \cos\phi \tag{57}$$

$$L/W = \frac{\rho_0^2\,(1 - R\theta)\,C_i^2\,V_i^2}{2p^2} + \sqrt{1 - \frac{C_i^2}{V_i^2} - 2\,\frac{C_i\Delta C_i}{V_i^2}} \tag{58}$$

Thus the effect of the true speed change is due to both an effect on the path angle and also to a curvature of the path. The magnitude of these effects for V_i = 80 m.p.h., C_i = 1,080 ft. per min., h = 10,000 ft., in standard atmosphere changes L/W by 0.0005 per cent because of curvature of the flight path and by 0.021 per cent because of inclination of the flight path.

Solar Heating of Bomb Cable

The effect of solar radiation on the measured air speed has been investigated as a possible source of error. The measured air speed is lowered by the increase in temperature in the cable. Consider the equation of isostasy

$$dp = -\rho g\,dh \tag{59}$$

where dp is the difference between the pressures at the bomb and at the air-speed indicator, ρ is the density of the ambient air at the average conditions, and dh is the vertical distance corresponding to dp. Inside the bomb cable, the same equation holds, but, if the temperature has risen,

$$dp_c = -\rho_c g\,dh \tag{60}$$

The static pressure error will then be

$$dp_c - dp = -g\,dh\,(\rho_c - \rho_a) \tag{61}$$

Resulting from this discrepancy is a measured velocity error. The air-speed indicator operates approximately according to the equation,

$$q = {}^1/_2\,\rho V^2 \tag{62}$$

from which is obtained

$$dV = dq/\rho V \tag{63}$$

Now the static pressure difference created by the heated

* Credit for this derivation of Eq. (51) should be given to Dr. F. H. Clauser.

air in the cable is identically dq. Further, since it is an excellent approximation to say

$$d\rho/\rho = -dT/T \tag{64}$$

the velocity error will be

$$dV = (1/\rho V)\ [-g\ dh\ (\rho_c - \rho_a)] \tag{65}$$

$$dV = g\ dh\ dT/VT \tag{66}$$

This can be written as

$$(dV/dh)/(dT/T) = g/V \tag{67}$$

For a typical case, use $dh = 80$ ft. and $V_i = 80$ m.p.h. at 10,000 ft. Estimates of the temperature rise to be expected show $dT/T = 0.01$ will be a value as great as can ever be expected. Then dV can be calculated to be 0.2 m.p.h., which is negligible.

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